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PREPARATIONS FOR TESTING A FULL SCALE OH-6A ROTOR SYSTEM WITH HHC INSTALLED

by

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June, 1995

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In the 1970's and 1980's Hughes Helicopters, now McDonnell Douglas Helicopters, in conjunction with the U. S. Army and NASA developed and flight tested the first successful Higher Harmonic Control system for the purpose of reducing helicopter vibration levels. In addition to the reduced vibration levels interesting benefits were also noticed in the areas of acoustic signature and helicopter power requirements. For many years the reduced power requirements could not be explained and therefore were downplayed and not publicized. In the past three years research at the Naval Postgraduate School has been conducted regarding this phenomena and it has been sufficiently explained. The next step in the process is to mount a full scale OH-6A main rotor system on a test stand and conduct testing in order to validate the earlier documented power savings. This thesis documents efforts that have been made in 1994 and 1995 towards accomplishing that goal.

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ABSTRACT

In the 1970's and 1980's Hughes Helicopters, now McDonnell Douglas Helicopters, in conjunction with the U.S. Army and NASA developed and flight tested the first successful Higher Harmonic Control system for the purpose of reducing helicopter vibration levels. In addition to the reduced vibration levels interesting benefits were also noticed in the areas of acoustic signature and helicopter power requirements. For many years the reduced power requirements could not be explained and therefore were downplayed and not publicized. In the past three years research at the Naval Postgraduate School has been conducted regarding this phenomena and it has been sufficiently explained. The next step in the process is to mount a full scale OH-6A main rotor system on a test stand and conduct testing in order to validate the earlier documented power savings. This thesis documents efforts that have been made in 1994 and 1995 towards accomplishing that goal.

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I. INTRODUCTION

A. GENERAL

Dynamics play a major role in the design and development of the helicopter. Vibratory characteristics have been the decisive factor in many helicopter acquisition competitions and have been the cause for cancellation of others. Low vibratory levels are desirable for many reasons: increased airframe life, increased component fatigue life, passenger comfort, crew comfort and crew fatigue. Since the mid 1950's it has been the goal of the Defense Department and industry to decrease helicopter vibration levels to the equivalent of fixed wing levels, approximately 0.02 g's. Figure 1 shows the trend of helicopter vibration levels from 1955 through 1985. Note that while there has been a dramatic decrease in the vibration levels, the decrease goes asymptotic at approximately 0.1 g.

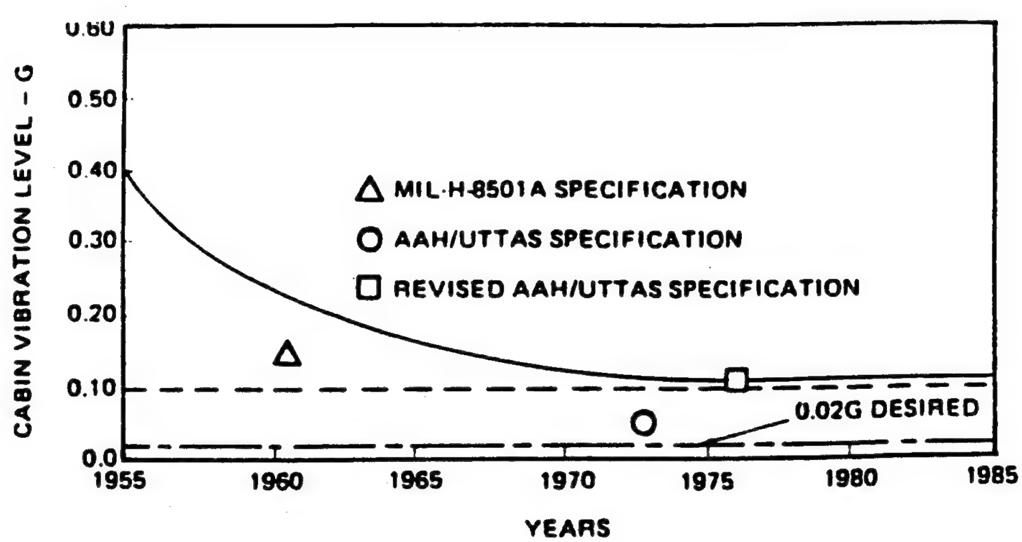

Figure 1. Trend of Helicopter Vibration Levels (From Ref. 1)

The overwhelming majority of vibration control schemes are passive concepts such as vibration absorbers and isolators. A passive vibration control device treats the vibratory loads after they have been generated and entered the airframe. While passive devices may be very effective at decreasing vibrations at specific locations, they do very little for the overall airframe. The vibratory energy has already entered the airframe and it must go somewhere; it will usually manifest itself where it is least desired. The key to controlling helicopter vibrations, and to likewise reap the benefits of low vibration levels, is to attack the vibratory loads with an active control system at their primary source, the rotor disk.

Higher Harmonic Control (HHC) is an active computer controlled vibration suppression system which alters the aerodynamic loads on the rotor and therefore reduces the vibratory forces and moments which enter the airframe. It accomplishes this by continuously monitoring the vibrations created by the rotor and suppresses them through high frequency feathering of the rotor blades. Two schemes currently exist for the implementation of HHC, direct HHC and individual blade control (IBC). Direct HHC works by oscillating the stationary swashplate to modify the airloads of the rotor blades collectively. With IBC each rotor blade has its own actuator which is used to control the pitch of each rotor blade individually. To date there have been several wind tunnel studies and four flight test programs dedicated to the understanding and possible implementation of HHC. The majority of that work has been done with direct HHC. Research is currently ongoing in industry with IBC. In addition to the expected results in the area of vibration reduction, there are several other potential benefits associated with HHC:

1. Reductions in power required
2. Reductions in acoustic signature
3. Possible use as a blade deicing mechanism

B. SCOPE

The purpose of this thesis is to document efforts made since July, 1994 to initiate a research program to study the effects of HHC on helicopter hover and low airspeed power requirements, external acoustic signature, vibration reduction capability, and possibly its use as a rotor blade deicing mechanism. The research program is a cooperative effort between the Naval Postgraduate School (NPS), the U. S. Naval Academy (USNA) and McDonnell Douglas Helicopters (MDHC). Recently, SATCON Technology, a controls engineering firm, has been awarded SBIR's from both the Army and Navy to study the feasibility of installing HHC in UH-60 and SH-60 helicopters. They are very interested in the proposed research and have contacted all of the included parties in an attempt to also be included. The intent of the research initiative is to mount a fully instrumented, HHC equipped OH-6A main rotor system on the rotor test facility located at the USNA and acquire quantitative vibration, acoustic signature and power required data. Specifically, this thesis will address efforts to acquire a rotor system and present preliminary plans for the work required to complete the research program.

II. BACKGROUND

A. HELICOPTER VIBRATIONS

Helicopter vibration is primarily the oscillatory response of the helicopter airframe to rotor hub forces and moments. Reference 2 offers a very good discussion of the subject. There are other sources of helicopter vibration, but the rotor system is the major contributor. In steady state forward flight, the periodic forces at the blade root is transmitted to the airframe producing a periodic vibratory response. Therefore, the vibratory response of the helicopter fuselage can be characterized as a harmonic response to the rotor system. The frequency of the forcing function is primarily at the one per revolution (1P) frequency and the n per revolution (nP) frequencies. (n is the number of blades). Other contributions are made by higher harmonics of the nP. The vibration amplitudes are generally low in a hover and increase with increasing forward flight speed to high levels at the maximum forward flight speed. There is also a high level of vibration during transition to forward flight and transition from forward flight to a hover due to the interaction of the rotor blades with the shed vortices of the preceding blade, known as blade-vortex interaction (BVI).

One per revolution vibrations are caused mainly by aerodynamic and inertial dissimilarities between the blades. The aerodynamic and inertial dissimilarities can normally be eliminated through tracking and balancing of the rotor system. The inertial dissimilarities can be eliminated by the addition of balance weights and the aerodynamic dissimilarities can typically be eliminated by adjusting trailing edge trim tabs or pitch rods.

N per revolution vibrations are due to the higher harmonic loading of the rotor. The sources of the higher harmonic loading are the rotor wake, the effects of advancing blade compressibility and retreating blade stall. In a hover where the aerodynamic environment is nearly axisymmetric the vibration level is low. The only sources of higher

harmonic loading in a hover are the small asymmetries due to the aerodynamic interference with the fuselage and other rotors. During transition to forward flight there is a peak in the vibration level due to the wake induced loads on the rotor. At the low advance ratios associated with transition, the drag of the fuselage is low enough so that the rotor disk incidence angle is small and the tip vortices remain close to the tip path plane. At the same time the advance ratio is high enough so that the rotor blades encounter the tip vortices of the preceding blade. This blade-vortex interaction produces an impulse type loading which is the source of significant higher harmonic blade loading. As speed is increased the tip path plane is tilted forward to provide propulsive force, the wake is convected away from the helicopter and wake induced vibrations decrease. At still higher speeds the vibrations begin to increase once again due to the higher harmonic loading created by advancing blade compressibility effects and retreating blade stall. Figure 2, from Reference 3, illustrates the varying airloads in forward flight as well as the convention used to measure rotor azimuthal position.

B. ROTOR AS A FILTER

Regardless of the type of rotor system (articulated, rigid, teetering, etc.), there is some method incorporated into the rotor hub design to relieve the flapping and lead-lag bending moments at the blade root. However, flapping and lead-lag shear forces still exist at the blade to hub attachment point. These forces sum at the rotor hub and form the vibratory forces that are transmitted to the fuselage. Many of the root shear summations are zero. Reference 2 gives a good explanation and derivation showing that the forces from all of the blades will exactly cancel at the hub except for those at harmonics of the nP. Therefore, in effect the rotor acts as a filter, transmitting to the fuselage only those forces which occur at integer multiples, or harmonics, of the nP. This result is based on the assumption that the rotor system is symmetric, that all of the blades are identical and that they all have the same periodic motion. While these assumptions

$$U_T = \Omega r + V\sin\psi$$

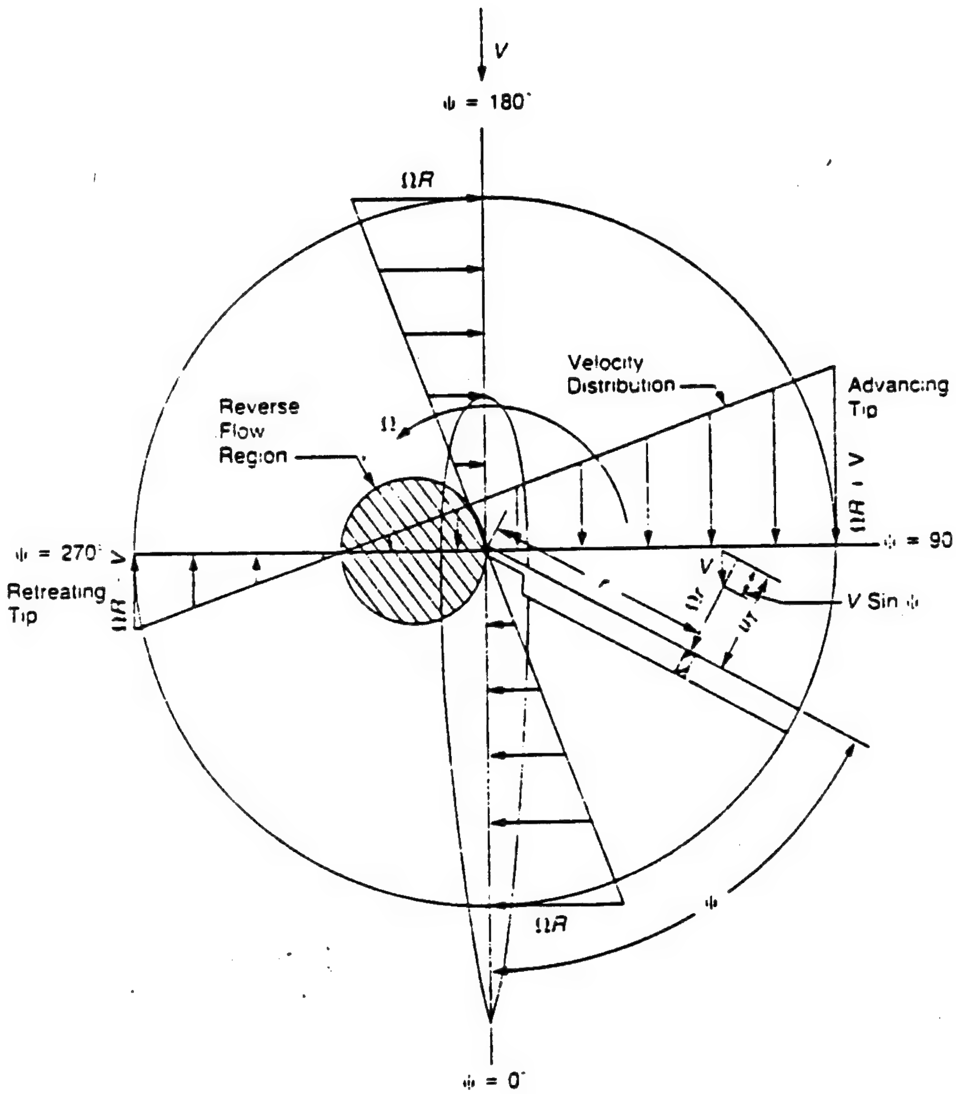

Figure 2. Inflow Distribution in Forward Flight (From Ref. 3)

are not always perfectly true, [illegible] harmonics still dominate the vibratory content of real rotor systems. This filtering process makes the task of vibration reduction or avoidance much easier in that fewer vibratory frequencies need to be considered. Table 1 lists the forces and moments present in the rotating frame along with their frequency components and the corresponding forces and frequencies that are filtered through to the non-rotating frame.

Rotor force and frequency (rotating frame)	*Fuselage force and frequency (non-rotating frame)*
vertical shear at n per rev	thrust at n per rev
lagwise moment at n per rev	torque at n per rev
in-plane shear at n +/- 1 per rev	fore/aft and lateral forces at n per rev
flapwise moment at n +/- 1 per rev	pitch and roll moments at n per rev
feathering moments at n per rev	collective control system forces at n per rev
feathering moments at n +/- 1 per rev	cyclic control system forces at n per rev

Table 1. Transmission of helicopter vibration from the rotating to non-rotating frame. (After Ref. 2)

As an example, consider the case of the OH-6A helicopter which has a four bladed, articulated rotor system:

- 3P and 5P flapwise blade root shears in the rotor result in 4P pitching and rolling moments in the airframe.
- 4P flapwise blade root shears in the rotor result in 4P vertical forces in the airframe.
- 3P and 5P chordwise blade root shears in the rotor result in 4P longitudinal and lateral forces in the airframe.
- 4P chordwise root shears result in 4P yawing moments in the airframe.

Since the normal rotational speed of the OH-6A rotor is 483 rpm the 1P exciting frequency is approximately 8 cycles/second and the 4P is approximately 32 cycles/second.

The higher harmonics of the of the 4P (8P, 12P, 16P, etc.) are computed in the same manner. However, the amplitudes of the higher harmonic forces are much smaller than the nP and are often ignored in analysis.

C. VIBRATION ALLEVIATION

Much time and effort is expended during the design and development of a helicopter to analyze its vibratory characteristics and alleviate the vibrations. The airframe is normally designed to avoid resonances with the 1P and nP of the rotor rotational frequency. However, components mounted within the airframe, such as avionics and crew stations, must usually be isolated in order to help prevent damage and crew fatigue. The most common way of accomplishing this is through the use of vibration isolators or absorbers which are tuned for a specific piece of equipment and for specific frequencies. Some helicopter designs utilize more elaborate means, such as the vibration suppression system in the Bell AH-1W SuperCobra or the nodal beam absorber in the Bell 222. Westland has developed an elaborate hydraulic force canceling system which utilizes active feedback control. In some cases these types of suppression systems alleviate the vibration in one component at the expense of another, actually causing damage to other components. These types of vibration suppression are adequate for specific applications, but the vibratory forces have already entered the airframe and can therefore cause damage throughout the airframe. The most effective way to alleviate the inherent vibrations and vibratory loads of the helicopter is to attack the problem at its primary source, the rotor.

D. HIGHER HARMONIC CONTROL

As previously stated, the major source of helicopter vibrations is the rotor system. These vibrations occur essentially at one frequency (1P) and the harmonics of that

frequency (nP). They can be suppressed with an active vibration suppression system such as Higher Harmonic Control (HHC). As the name implies, HHC works by modifying the aerodynamic loads of the rotor blade at the higher harmonics of the nP. For the purposes of this discussion, HHC is a computer controlled, active vibration suppression system which counters the nP vibrations induced by the main rotor. It does this by continuously monitoring rotor induced vibrations, and counters these inputs by high frequency feathering of the rotor blades. The feathering is at integer multiples of the nP, hence the name. Two methods currently exist for implementing HHC, direct HHC and Individual Blade Control or IBC. This paper will be restricted to direct control HHC.

Direct HHC works by superimposing nP swashplate motion on the basic collective and cyclic control inputs through the existing helicopter control system. For clarity, Figure 3, from Reference 3, shows a schematic of a generic helicopter control system. From the figure it can be seen that the swashplate transfers control motion from the non-rotating to the rotating frame. In the direct HHC method the existing stationary swashplate is oscillated in the collective, longitudinal and lateral directions by actuators that are fixed to the airframe. The amplitudes are small and the oscillation frequency is at the nP. When properly phased, these oscillations create incremental airloads on the blades which cancel the normal vibratory loads encountered in flight. The three modes in which the swashplate can be moved are described as follows:

- collective mode- refers to the collective movement of the swashplate in the vertical direction so that all of the rotor blades receive the same pitch change simultaneously.
- lateral mode- refers to the lateral tilting of the swashplate.
- longitudinal mode- refers to the fore and aft tilting of the swashplate.

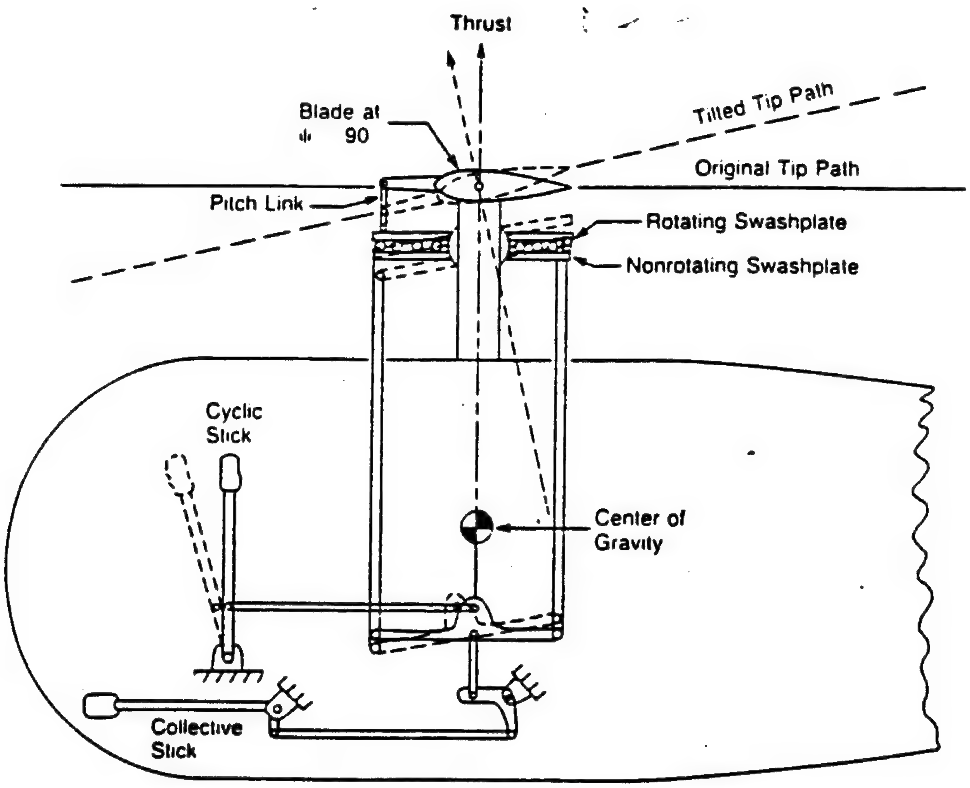

Figure 3. Schematic of Helicopter Control System (From Ref. 3)

Past testing indicates that optimal vibration reduction is accomplished by introducing the oscillations in combinations of the three modes. Referring to section B of this chapter, the filtering process of the rotor also works in reverse. An nP longitudinal or lateral tilt of the swashplate results in (n-1)P and (n+1)P blade feathering motion. By varying the amplitude and phase of the HHC input, any combination of (n-1)P, nP and (n+1)P blade feathering may be obtained.

E. HUGHES HELICOPTERS, Inc. OH-6A HHC PROGRAM

In 1976 Hughes Helicopters, Inc. teamed with the U. S. Army and NASA in an effort to develop a flightworthy HHC vibration control system. The system was successfully flight tested in 1982 through 1984. While it was realized that HHC could provide numerous benefits in addition to vibration reduction, it was felt that the

pursuance of too many objectives at one time could be counter productive. For that reason the primary objective of the Hughes program was to design and test a system to minimize the 4P vibration of the OH-6A fuselage. The purpose of this section is to describe the overall HHC system concept used during the flight test as detailed in Reference 1.

The aircraft used for the Hughes program was an OH-6A. The OH-6A is a four bladed light scout helicopter. It utilizes an articulated main rotor system which incorporates lead-lag, flapping and feathering hinges. The main rotor rotates at 483 RPM. Therefore the 1P is approximately 8 Hz and the 4P is approximately 32 Hz. The standard OH-6A utilizes a reversible control system which makes it unsuitable for use with HHC due to the amount of control feedback the pilot would feel in the cockpit. For that reason an OH-6A that had been previously used to develop a 1500 psi hydraulic boost system for the primary controls was baled to Hughes for the program.

The primary elements of the Hughes HHC system were:

1. acceleration transducers to sense the vibratory response of the fuselage
2. a higher harmonic blade pitch actuator system
3. a flightworthy microcomputer
4. an electronic control unit (ECU)

Briefly, the system operated as follows. Tri-axial accelerometers mounted beneath the pilots seat sensed vertical, lateral and longitudinal vibrations and passed these signals to the ECU. The ECU converted these signals into an electronic format which could be read by the computer. In the conversion the ECU separated the sine and cosine components of the 4P signal. The computer analyzed the input signals and determined the amount of blade feathering required to cancel the vibration. This information was sent back to the ECU in computer format. The ECU would then convert this information to 4P analog signals which were the electrical input to the three high frequency servos that were used to drive the stationary swashplate collectively and in pitch and roll. The process was

repeated approximately every two rotor revolutions. The system was evaluated with both open and closed loop flight test programs.

F. HUGHES HELICOPTERS, Inc. HHC SYSTEM DESCRIPTION

Designing and incorporating an HHC system into an existing helicopter was a challenging task and called for clear, well thought out design objectives. Following is a partial listing of the more important design objectives, taken from Reference 1, with a brief justification of each, and a brief description of the HHC system as it was installed on the OH-6A.

In designing the HHC system for the OH-6A, Hughes had four primary design objectives:

1. To locate the HHC actuators in the stationary portion of the control system. Doing this avoided the need to generate 3P, 4P and 5P signals since any combination of those signals can be produced by the proper phasing of 4P signals to the stationary swashplate in collective, pitch and roll. Also, by placing the actuators in the stationary system there is no need for a rotating hydraulic slip ring and manifold assembly.
2. To design the system for one goal only, vibration reduction. The program was a proof of concept demonstrator; it was known that there were other possible benefits from the implementation of HHC but it was felt that to pursue too many objectives at one time would complicate the issue and be counterproductive.
3. To provide an HHC system that was completely independent of the primary control system. In that way the HHC signals were superimposed on the primary control signals. This objective offered many benefits. By being independent of the primary control system, the HHC system had little effect on the basic rotor trim in flight and it also allowed complete helicopter control to

revert to the basic primary control system in the event of an HHC system failure. Also, by being completely independent, the HHC system and the primary control system could utilize separate hydraulic systems that best suited the needs of each. Finally, the HHC actuators could be located where they would be most effective. This last point is very important because the HHC actuators need to be located where they will be reacted by a high impedance in order to minimize lost motion from the actuator output due to control system flexibility and freeplay.

4. Isolate the 4P signals from the accelerometers by analog means. Initially it would seem logical to use a Fast Fourier Transform (FFT) to isolate the signals. However, the problem with FFT is the record length that is required and the record length was limited in this case by the sampling rate. For this system an electronic analog technique was applied that precluded the need for FFT methods and provided an essentially continuous signal.

Modifying an existing OH-6A to accommodate a HHC system established a number of challenging requirements which are summarized below, from Reference 1:

1. Development of high bandwidth HHC servo actuators.
2. To update the primary flight control system to permit high fidelity blade feathering
3. To work within the existing helicopter framework
4. Development of an adequate HHC controller

1. HHC Actuator Design

The actuators replaced existing drive links between the mixer assembly and the stationary swashplate as shown in Figure 4. The actuator design was driven by the

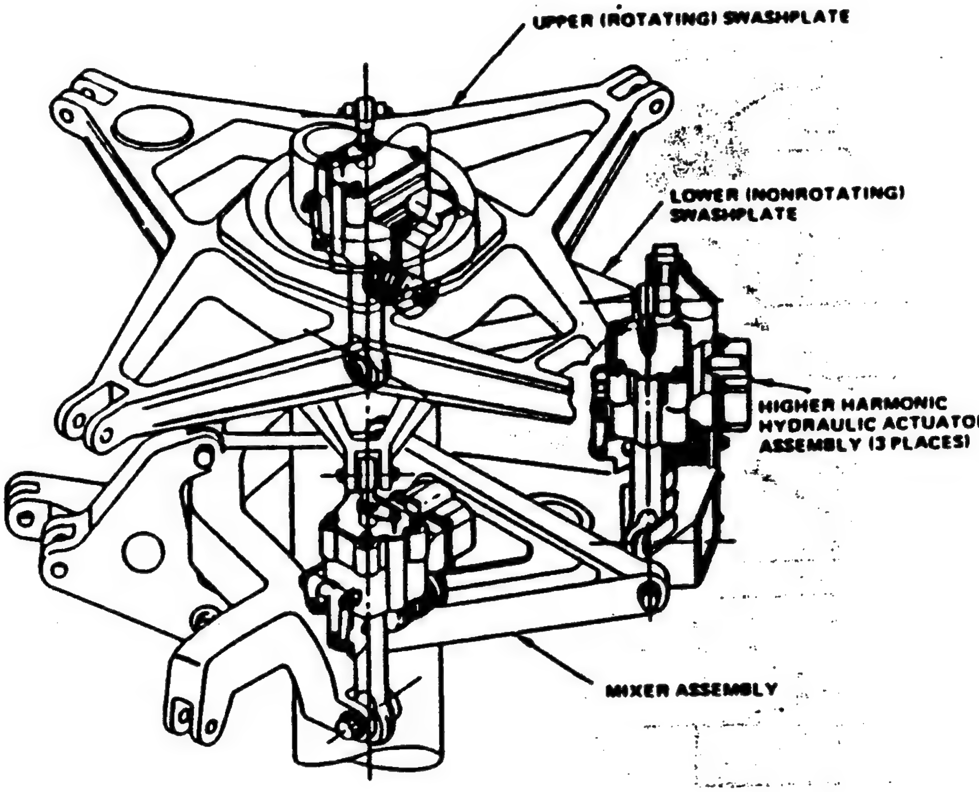

Figure 4. Swashplate actuator installation (From Ref. 1)

frequency response requirements. Piston area, drill passage diameter, seal friction and electro-hydraulic servo valve characteristics were all optimized to enhance the installed frequency response of the actuator. The actuators, designed and manufactured by MOOG, Western Development Center, were designed to have a total collective blade angle authority of two degrees or ± 0.20 inches of stroke. The usable frequency response was approximately 90 Hz for the installed actuator at command amplitudes of one degree of collective authority. Hydraulic operating pressure of the actuators was 3000 psi. A center-driving lockout device was incorporated to drive the actuator to the neutral position in the event of an actuator failure. Hydraulic power for the HHC servos was provided by a Sperry-Vickers variable displacement pump which operated at 2800 RPM. A Bertea integrated manifold/reservoir provided the distribution network needed to filter, cool, accumulate and route the hydraulic fluid to the actuators.

2. Primary Flight Control System

During the initial checkout of the high frequency HHC actuators it was determined that the existing OH-6A mechanical flight control system was incapable of transmitting the high frequency feathering motion to the rotor blades because of excessive freeplay and flexibility of certain control system components. A test program to isolate the principal sources of lost motion revealed that there was a considerable amount of freeplay in all three control axes. However, true freeplay
(i.e. zero stiffness) contributed less to the freeplay than did bearing, bushing, bolt and bellcrank flexibilities. Through the use of precision tolerance bearings, bolts, bushings, metal to metal rod end bearings and redesigned mixer components, an 80% reduction in total system freeplay was realized along with a 90% increase in system stiffness.

3. System Controller

At the heart of the HHC system is the controller and the control algorithm. In a direct HHC system the controller attempts to reduce vibrations in a measured response, usually vibrations measured in the vicinity of the pilots seat. The controller is responsible for sensing fuselage vibrations, computing the appropriate values for the transfer matrix and sending the appropriate signals to the HHC actuators for collective, lateral and longitudinal swashplate excitation.

The Hughes HHC controller model was based on the following equation,

$$Z_i = Z_{oi} + T_{ij} \boldsymbol{u}_j \tag{2.1}$$

Equation 2.1 assumes a linear transfer between the command input and the fuselage vibrations. The equation states that the system response 'Z_i' consists of a baseline response 'Z_{oi}' plus a response which is related to the command input 'u_j' by the transfer matrix 'T_{ij}'.

Reference 4 gives a good overview of controller types. In general there are two types of controller models which can be utilized, a global model or a local model. The global model assumes that the control law is linear for the entire range of its control. The local model, on the other hand, assumes that the control law is linear about a current control value and is applicable even for non-linear conditions since the transfer matrix (T) is linearized about a current value and the swashplate excitation (u) is small.

The transfer matrix (T) and the baseline vibrations vector (Z_o) can be identified in two different ways, off-line or on-line. In the off-line method, the components of the transfer matrix and the baseline vibrations matrices are generally computed from wind tunnel testing or flight test and are considered invariant. Off-line controllers can be further classified as fixed gain or scheduled gain controllers. For the fixed gain controller the control law matrices remain fixed throughout the flight envelope. Scheduled gain controllers use predetermined matrices that are phased into use based on some aircraft flight parameter. Off-line controllers are suitable for use only with global control models. With the on-line method, the characteristics of the matrices are continuously updated. These controllers are also called adaptive controllers since the control gains vary with the flight conditions. On-line identification is applicable to both global or local models. There are many versions of this type of identification scheme, some update only the Z_o vector while others update both the T matrix and the Z_o vector. One popular method for updating and predicting the values of the matrices is the use of Kalman filters. Finally, many controllers use caution terms in the algorithm in order to prevent large discontinuities in the control laws between updates.

The control algorithm used by Hughes in their HHC program was developed by John Molusis of the University of Connecticut and is reported in Reference 5. It was considered both cautious and self adaptive due to the presence of caution terms to prevent large changes in the control inputs from one iteration to the next and the use of a Kalman filter to estimate the various parameters of the T matrix and Z_o vector at each iteration. The control inputs at each iteration were then based on an optimal solution of the model.

This approach allows the use of the algorithm without prior knowledge of the system being controlled and has the advantage of being transportable from one helicopter to another without doing extensive flight test to develop control derivatives and gains as a function of flight regime.

In order to facilitate further discussion, equation 2.1 is reiterated here.

$$Z_i = Z_{oi} + T_{ij}\boldsymbol{u}_j$$

In the OH-6A controller, Z_i was a 6x1 vector of the measured vibrations at the pilots seat with HHC on, Z_{oi} was a 6x1 vector of the measured baseline vibrations with HHC off, T_{ij} was a 6x6 matrix which related the change in vibration levels to HHC inputs and u_j was a 6x1 vector of the commanded HHC inputs. Although only three quantities are measured, (the longitudinal, lateral and vertical accelerations) the 4P component of each of these is separated into its sine and cosine elements, which gives the six elements for the vectors.

The Hughes HHC controller utilized the on-line method to identify the values of the Z_{oi} and T_{ij} matrices. During the open-loop flight testing the method chosen to initialize the Z_o and T matrices was a straightforward application of open-loop control inputs. First, baseline vibration levels were recorded and then open loop inputs were applied for each element of the control vector individually and the responses measured. In this way the respective columns of the T matrix could be determined for each input. Figure 5 is a schematic of this process. Note that there was a pause incorporated after each control application to allow the vibrations to stabilize before the measurements were recorded. Once the controller was initialized, a new estimate of Z_o and T were obtained at each iteration and the optimal controls were based on these calculations. Once initialized, the system operated as depicted in Figure 6. The flight conditions under which the controller was initialized did not appear to have a significant influence on controller operation.

During closed loop flight testing the controller initialized itself during the auto cal phase. Figure 7 is a strip chart of a typical engagement and operation of the HHC system

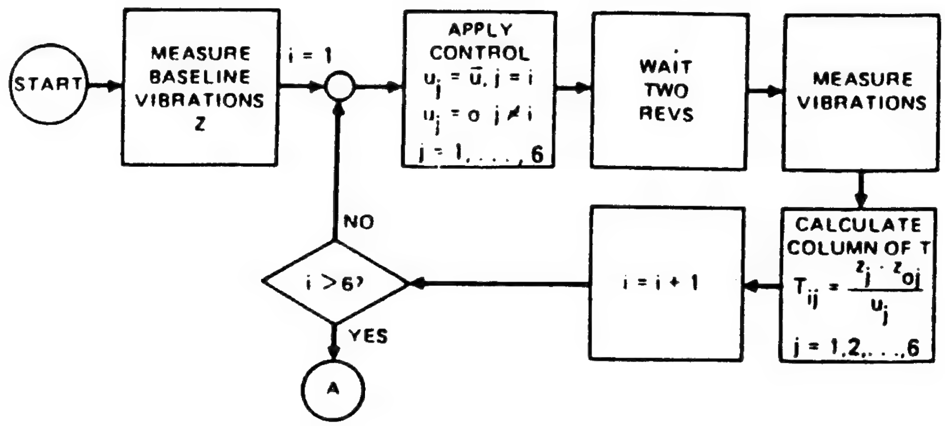

Figure 5. Schematic of Controller Initialization
(From Ref. 1)

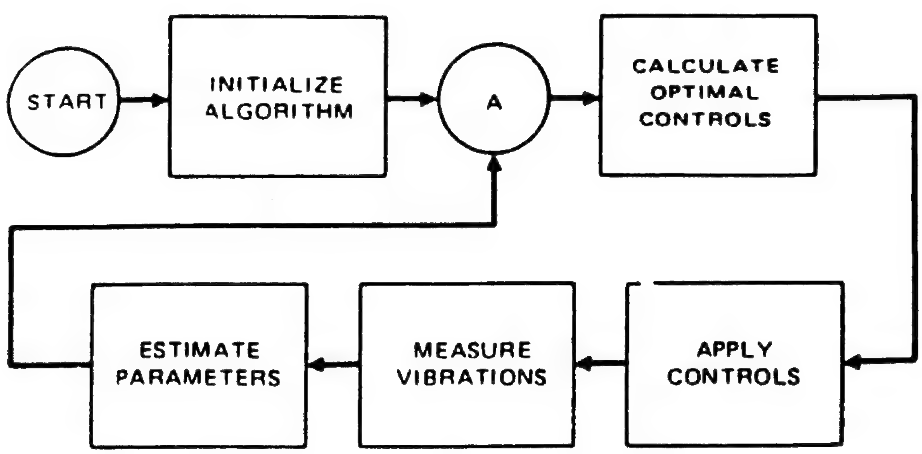

Figure 6. Schematic of Controller Operation
(From Ref. 1)

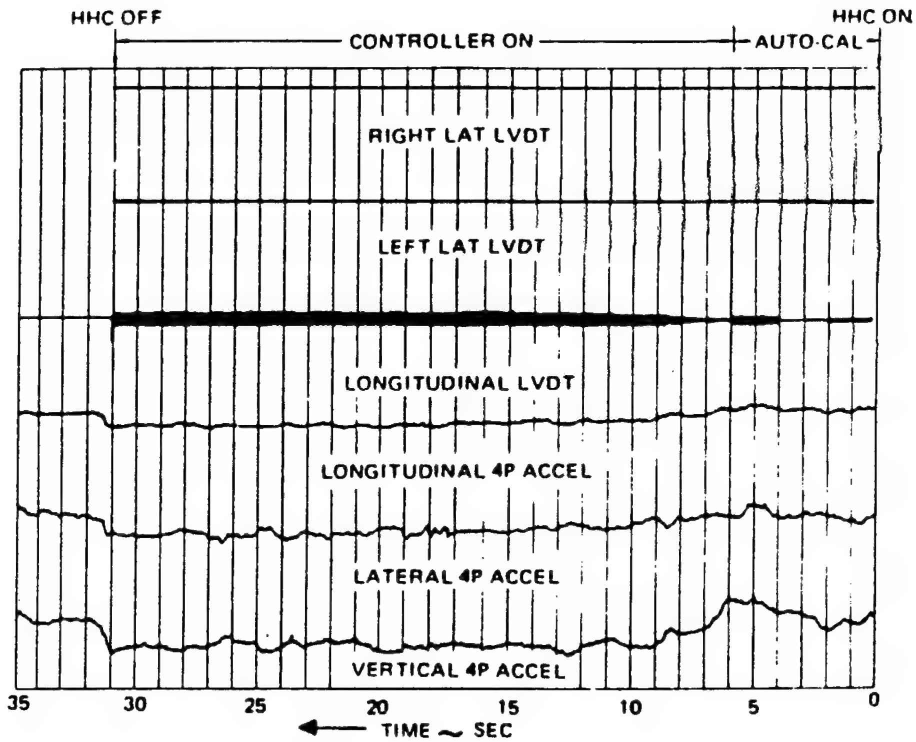

Figure 7. Closed Loop HHC Engagement (From Ref. 1)

at 60 Knots indicated airspeed (KIAS). In Figure 7, time increases from right to left. Engagement of the HHC system can be seen on the three LVDT traces during the initialization, or auto-cal phase. During the auto-cal phase each actuator was operated individually and the response was measured. From each individual channel response three elements of the T matrix were generated. Shown in the figure are the time histories of the three HHC actuators and the sine components of the 4P longitudinal, lateral and vertical accelerations measured at the pilots seat. The longitudinal trace appears noisy due to less stiffness in the longitudinal axis than in the other axes, therefore the longitudinal actuator was being overdriven. Once initialization was complete the closed loop controller had sufficient information to operate the system. Notice the gradual decrease in the vibration levels after the controller begins operation. This is evidence of the caution terms in the algorithm. Disengagement of the HHC system is readily apparent at the sudden jump in the vibration traces back to the baseline values.

G. FLIGHT TEST RESULTS

The purpose of the open loop flight testing was to obtain a data base for subsequent closed loop testing. The objective of the closed loop testing was to evaluate the performance of the HHC system in simultaneously minimizing the longitudinal, lateral and vertical vibrations measured at the pilots seat. Level flight testing, both open and closed loop, was conducted at a hover and from 40 to 100 KIAS in 10 knot increments. Several mission maneuvers such as coordinated and windup turns, approaches and flares and accelerations and decelerations were also performed. In both cases the procedure for gathering data was to stabilize on the desired airspeed, collect baseline HHC off data and then to engage HHC and record the HHC on data.

As previously explained, the HHC manual controller could be operated in one of three modes or in a combination of the three modes. The modes were the 4P longitudinal and lateral cyclic and the 4P collective modes. During open loop flight testing each mode

was operated independently at each airspeed throughout the tested envelope. At each airspeed, HHC blade angle motion was set for each given mode and a phase angle sweep was conducted from 0 to 360 degrees at 30 degree increments in order to determine the optimum input phase angle for maximum vibration reduction. Each open loop data run took between 10 and 15 minutes. The input phase was referenced to a specified azimuth position taken from a nominal zero position. A separate instrument recorded this position. It refers to the phase of the swashplate tilting in relation to the main rotor position. Note that for a four bladed rotor 360 degrees of phase corresponds to 90 degrees of rotation of the rotor.

As designed and installed on the OH-6A, the HHC system had a maximum ± 2.0 degrees of blade pitch authority. Preliminary wind tunnel testing indicated that this was probably more than would be required and an electronic limit was designed into the ECU which restricted the blade angle authority to ± 1.0 degrees. The open loop flight testing was flown using only half of that. Accounting for lost motion and component flexibility, the HHC inputs were approximately ± 0.33 degrees of blade pitch motion.

1. Vibration Reduction

Figures 8-12 show the effect of HHC lateral swashplate excitation on vibrations during open loop testing at 60, 70, 80, 90 and 100 KIAS respectively. In the figures, vertical and lateral vibration levels, as measured at the pilots seat, are plotted on the vertical axis in units of g's versus the phase angle of the commanded HHC input on the horizontal axis. Also plotted on the figures in dashed lines are the baseline vibration data taken for each particular airspeed.

Several things are common to all five of the figures. The general shape of all of the plots is the same and it is readily apparent that HHC can increase vibrations as well as decrease them. Note from Figures 8, 9 and 10 that the maximum amount of vibration that

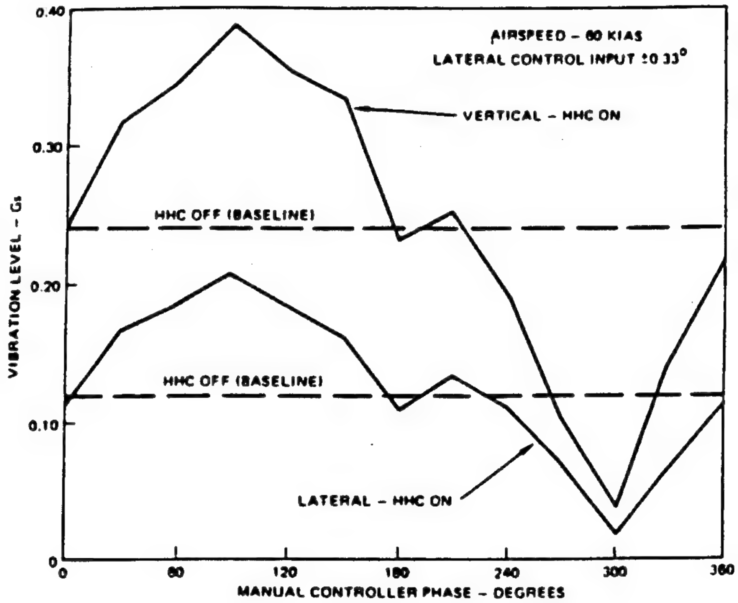

Figure 8. Variation of 4P vibration with input phase at 60 KIAS (From Ref. 1)

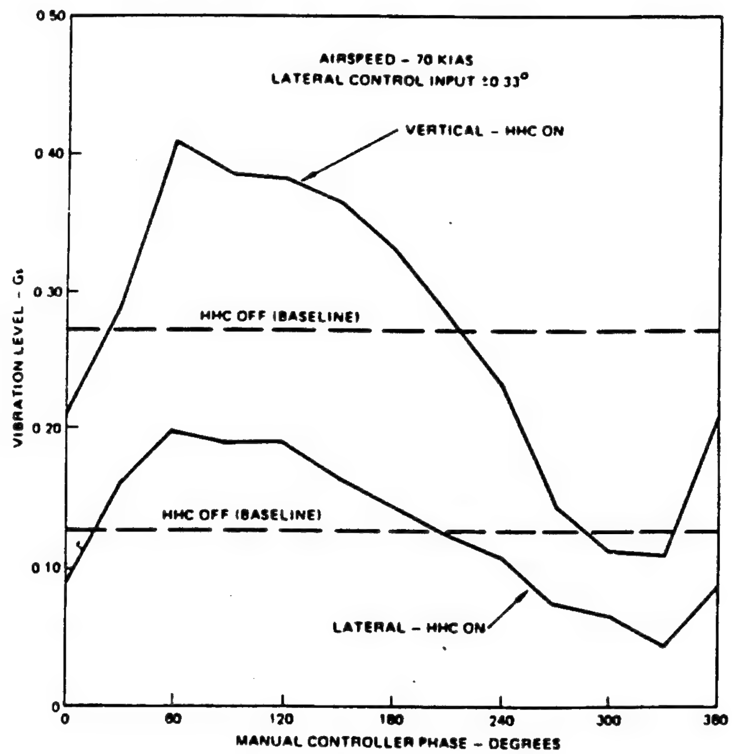

Figure 9. Variation of 4P vibration with input phase at 70 KIAS (From Ref. 1)

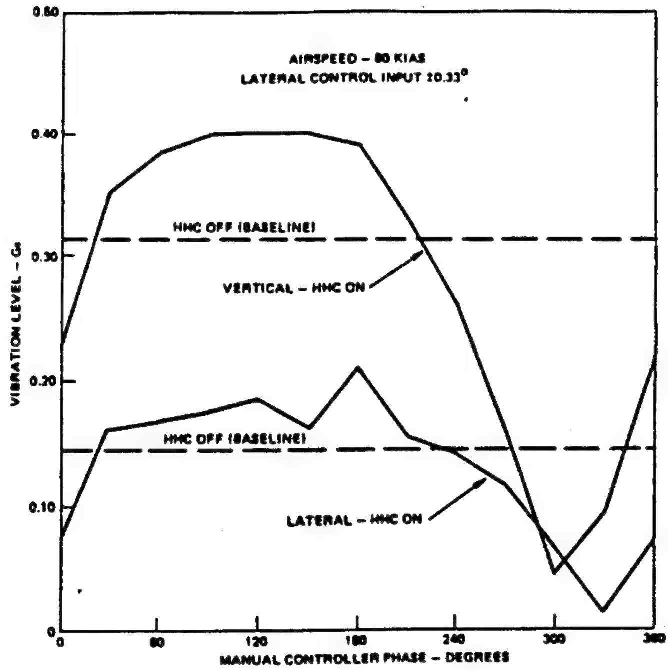

Figure 10. Variation of 4P vibration with input phase at 80 KIAS (From Ref. 1)

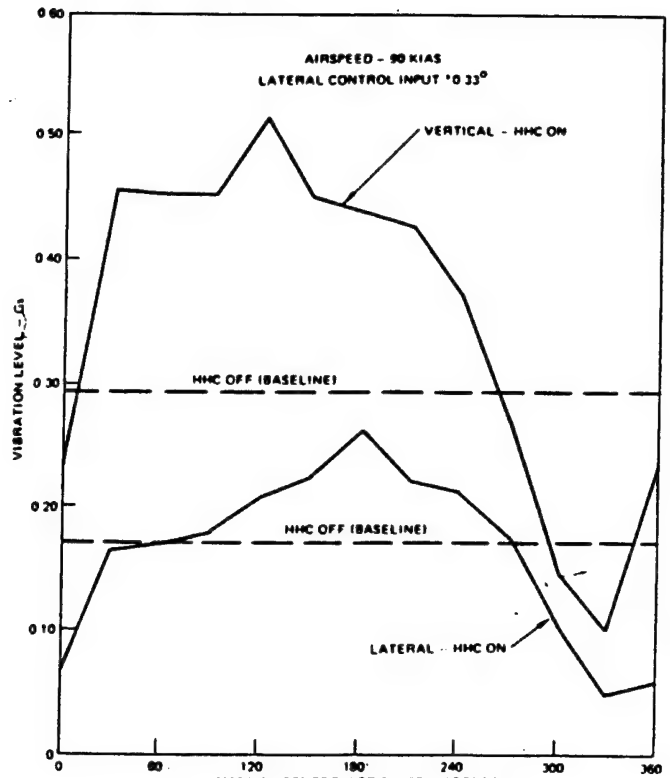

Figure 11. Variation of 4P vibration with input phase at 90 KIAS (From Ref. 1)

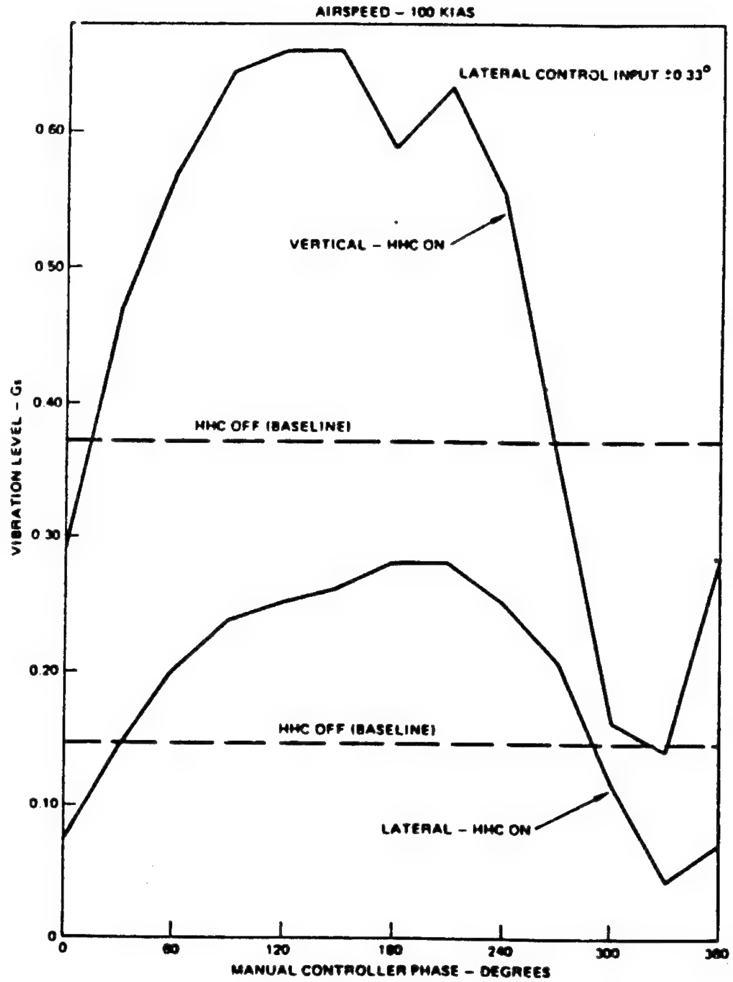

Figure 12. Variation of 4P vibration with input phase at 100 KIAS (From Ref. 1)

was induced into the airframe w[illegible] [illegible]pproximately 0.4 g vertically and 0.2 g laterally for 40,50, and 60 KIAS. In Figure[illegible] [illegible]1 and 12 these values jump to 0.51 g vertically and 0.26 g laterally for 90 KIAS and 0.66 g vertically and 0.28g laterally for 100 KIAS. The increase in maximum vibration levels at 90 and 100 KIAS were attributed to a combination of the main rotor excitation and the oscillatory impingement of the rotor wake on the tailboom. In general the minimum vibration occurred with an input phase angle of approximately 300 to 330 degrees. It is not surprising that this is approximately 180 degrees out from where the maximum vibrations occurred. On average, the maximum reduction in vibration levels from baseline was approximately 71% in the vertical axis and 73% in the lateral axis. These are significant reductions and were a result of HHC inputs in only one axis.

As previously stated the purpose of the closed loop testing was to evaluate the effectiveness of the HHC system in simultaneously reducing the vibrations in all three axes to a minimum. Therefore after system initialization, the full transfer matrix was utilized in obtaining the optimal solution to the control law governing equation, with the controller determining the proper input amplitudes and phase angles. Figures 13 through 15 show the effect of HHC closed loop operation on accelerations at the pilots seat during initial closed loop testing. In Figures 13 through 15 the Kalman filter had not yet been optimized. Shown is the fourth harmonic of the accelerations in all three axes plotted versus airspeed. It can be seen that HHC was successful at attaining significant reductions in the lateral and vertical vibration levels throughout the speed range tested. In the longitudinal axis significant reductions were obtained below approximately 65 KIAS. Also note that in the vertical axis the vibration reductions at the higher airspeeds are not as great as they are at the lower airspeeds. At these higher airspeeds there was no tendency for the controller to drive the swashplate towards the electronically set limits of the system, so it was concluded that the reduced effectiveness was not due to inadequate control authority.

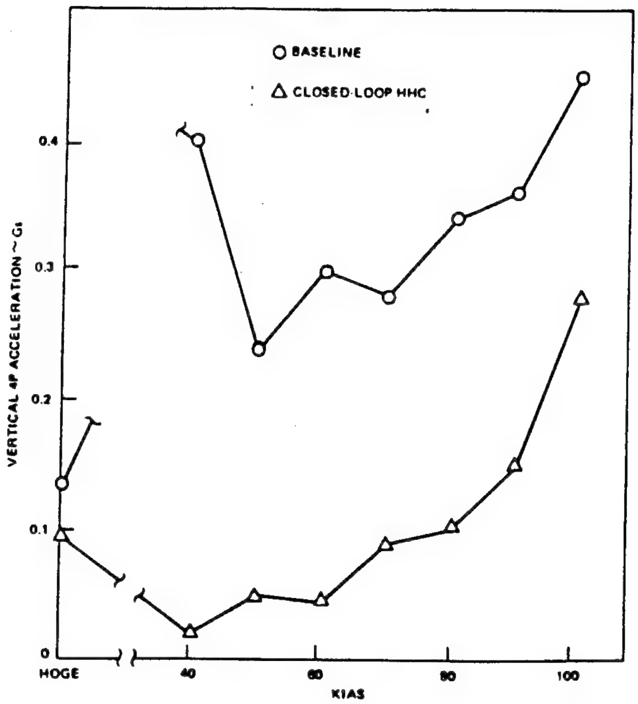

Figure 13. Closed loop 4P vertical pilot seat vibrations versus airspeed (From Ref. 1)

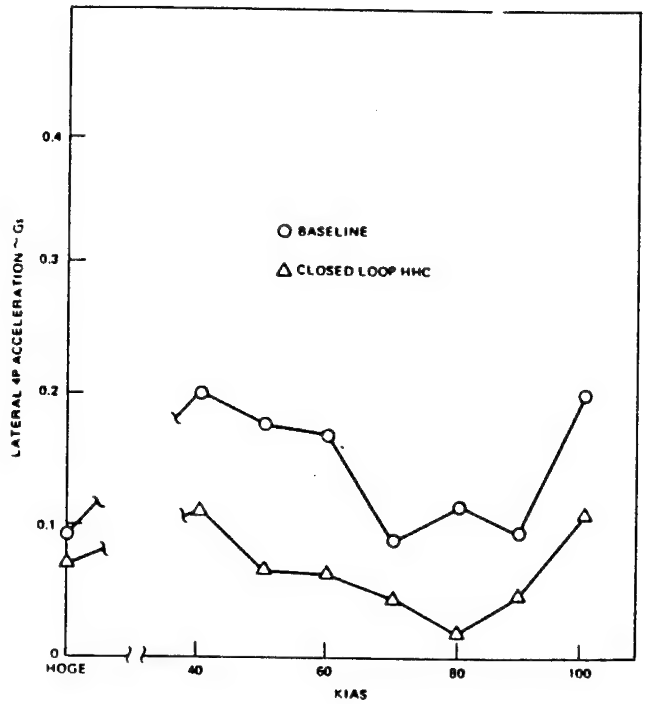

Figure 14. Closed loop 4P lateral pilot seat vibrations versus airspeed (From Ref. 1)

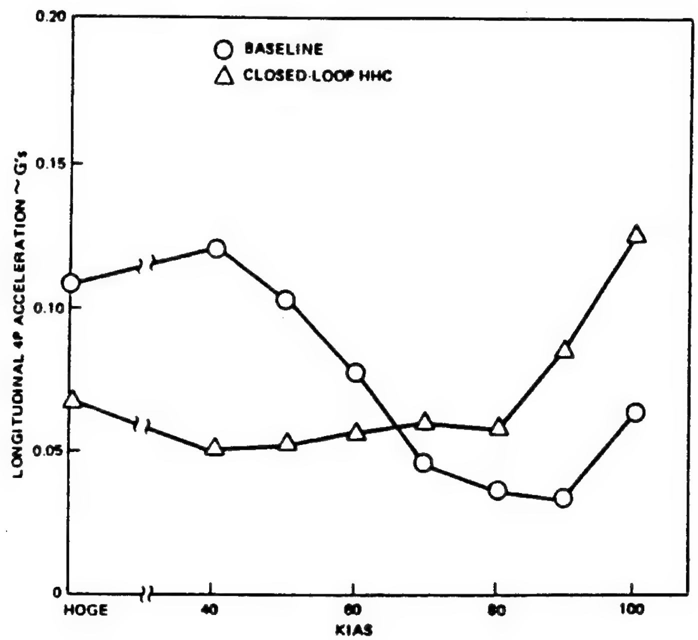

Figure 15. Closed loop 4P longitudinal pilot seat vibrations versus airspeed (From Ref. 1)

There are two other possible explanations for the reduced effectiveness. First, time history traces of the actuator motions indicated that the controller was commanding predominantly longitudinal inputs with only small lateral inputs. This contradicts data obtained in the open loop testing where it was seen that inputs in the lateral axis had the greatest effect on overall vibration level reductions. Furthermore, the longitudinal axis of the control system was the least stiff of the three axes and only one actuator drives the swashplate in the longitudinal direction vice the two for lateral and three for vertical. These factors point to the longitudinal axis as being the least effective for the purposes of HHC inputs.

The second possible explanation is the possibility of the system being nonlinear. If this were the case the controller would drive the vibrations to a local minima vice a global minima. In that case the controller would be very sensitive to the initialization conditions. For the data presented in Figures 13-15 the controller was initialized at the airspeed at which the data was taken.

Figure 16 shows the higher harmonic blade angles that were required at each of the data points plotted in the preceding three figures. The data in Figure 16 shows the blade feathering angles required for the third, fourth and fifth harmonics. Since the feathering angles were measured in the rotating system all three of the higher harmonic feathering angles could be measured directly. Notice that the blade feathering angles are less than 0.5 degrees through the speed range tested, which agrees with earlier wind tunnel testing.

Another indication of the effectiveness of HHC is the longitudinal and lateral mast bending moments. The OH-6A employs a stationary mast concept wherein all of the rotor loads and moments are transmitted directly to the fuselage through a static mast vice through the transmission as in other helicopters. Consequently it is relatively easy to measure the vibratory loads and moments that are being transmitted to the fuselage directly from the static mast. Figures 17 and 18 show the 4P longitudinal and lateral mast bending moments respectively versus airspeed for the OH-6A. Once again it can be seen that HHC has a beneficial effect throughout the speed range tested, but has less of an

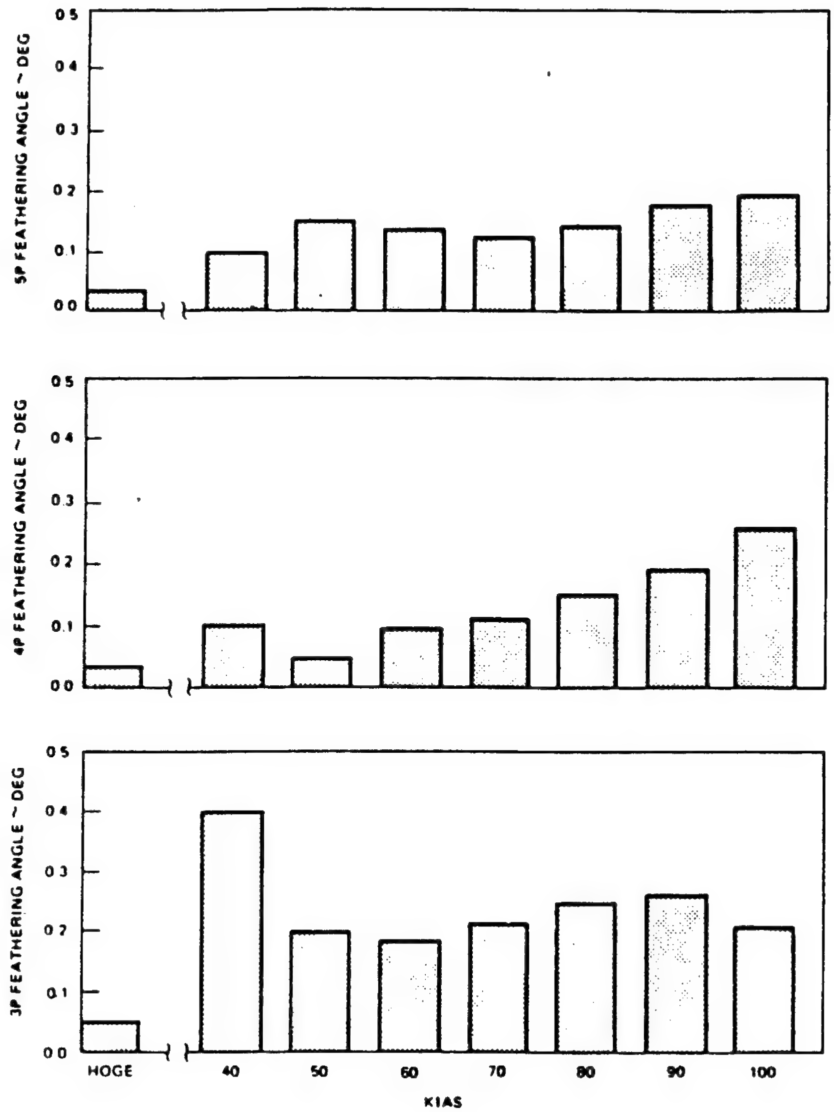

Figure 16. Closed loop 3P, 4P and 5P optimal blade feathering angle versus airspeed. (From Ref. 1)

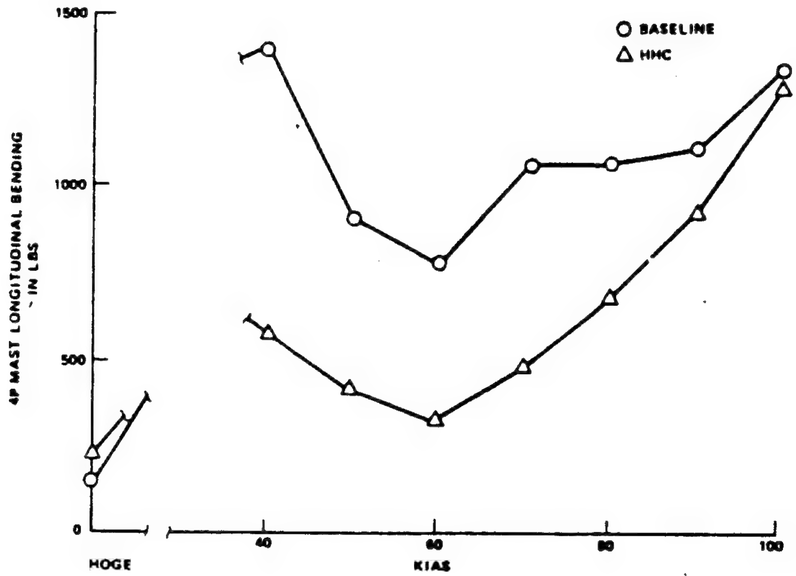

Figure 17. **Closed loop 4P longitudinal mast bending versus airspeed (From Ref. 1)**

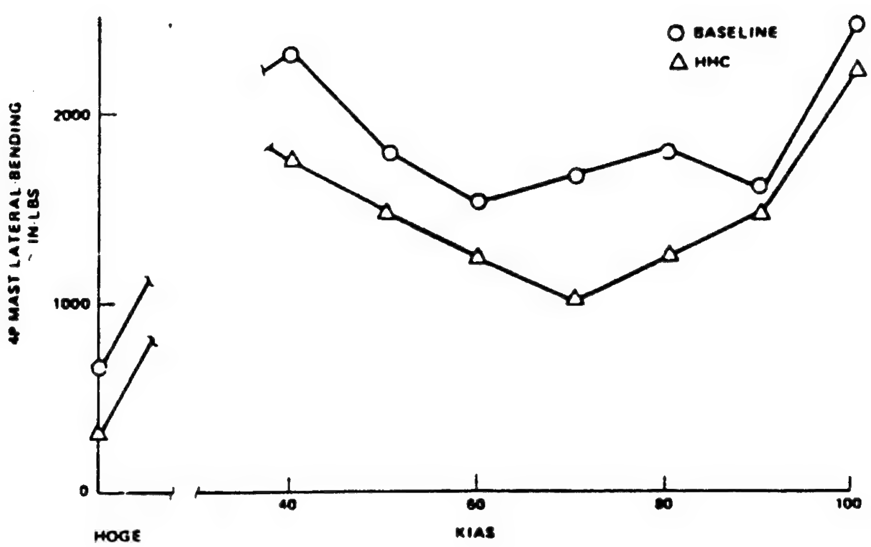

Figure 18. **Closed loop 4P lateral mast bending versus airspeed (From Ref. 1)**

effect at the higher airspeeds, which is consistent with the accelerometer data taken at the pilots seat.

Flight testing performed in 1984 utilized an improved control algorithm which reduced calculation time from 167 ms to 58 ms and resulted in a total update reduction time from 257 ms to 163 ms. Changes in the algorithm related to the way in which the computer performed calculations and in the way the Kalman filter determined the values for the gain vector. These changes resulted in tremendous overall improvement of the HHC system, especially at the higher airspeeds, and graphically indicate the importance of the controller to the overall performance of the system. Figures 19 through 21 show the results of closed loop testing with the new HHC algorithm.

2. HHC Effects on Power Requirements

An unexpected benefit from the use of HHC was a reduction in rotor power required. The primary purpose for the HHC investigation was to investigate its effects on vibration reduction and therefore the aircraft was not heavily instrumented for performance measurements, however engine torque pressure and main rotor shaft torque were measured and recorded. Figures 22 and 23 show the effects of HHC on the main rotor torque and engine torque pressure versus airspeed for the speed range tested.

Figure 24 shows the effect of a selected set of HHC inputs on power required during open loop testing. The data is more graphically presented in the polar plots. In Figure 24, from Reference 6, each data point indicates the power savings associated with HHC inputs applied at the corresponding phase angle. Note that the power savings are on the order of 10% for a hover. The power savings seems to be independent of the type of input but it does seem to be sensitive to the phase of the input. Figures 25 through 27, from Reference 6 present the same type of data for forward flight at 60, 80 and 100 KIAS. It can be concluded from Figures 22 through 27 that HHC has a beneficial effect on helicopter power requirements ranging from approximately 10% in a hover to

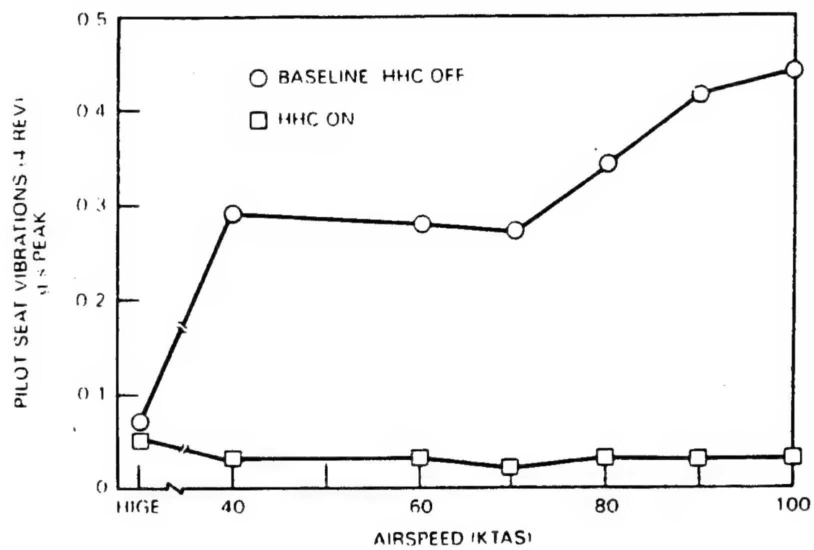

Figure 19. Closed loop 4P vertical accelerations versus airspeed with improved algorithm. (From Ref. 1)

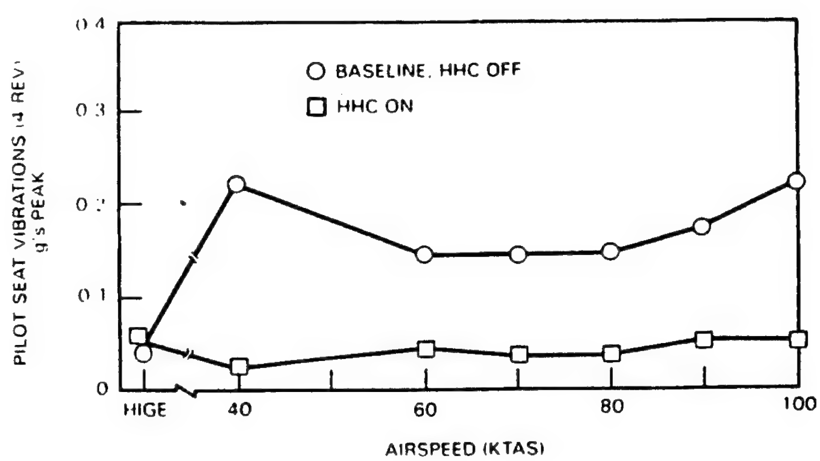

Figure 20. Closed loop 4P lateral accelerations versus airspeed with improved algorithm. (From Ref. 1)

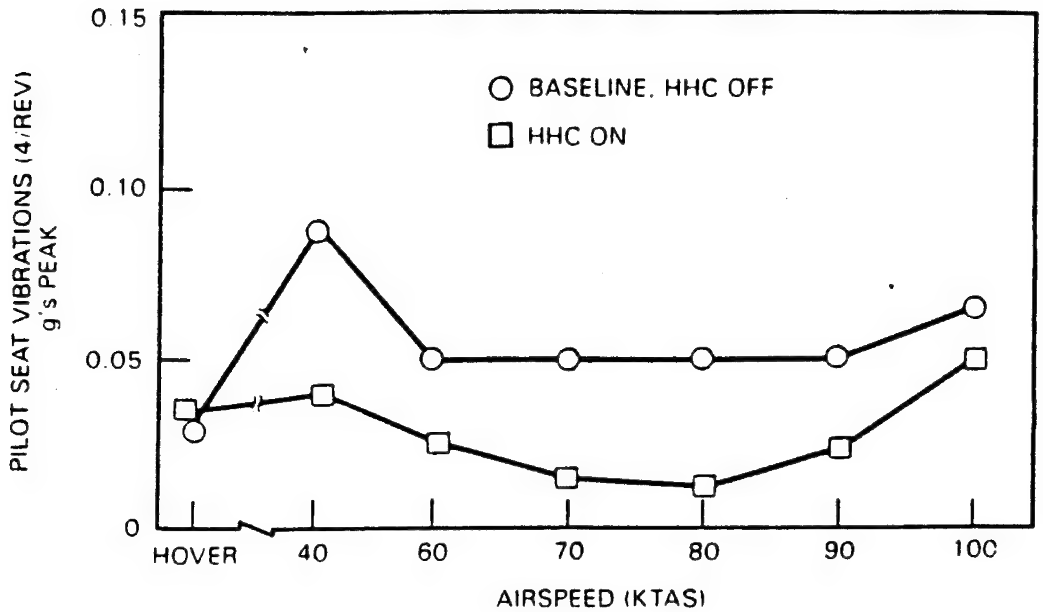

Figure 21. Closed loop 4p longitudinal vibrations versus airspeed with improved algorithm. (From Ref. 1)

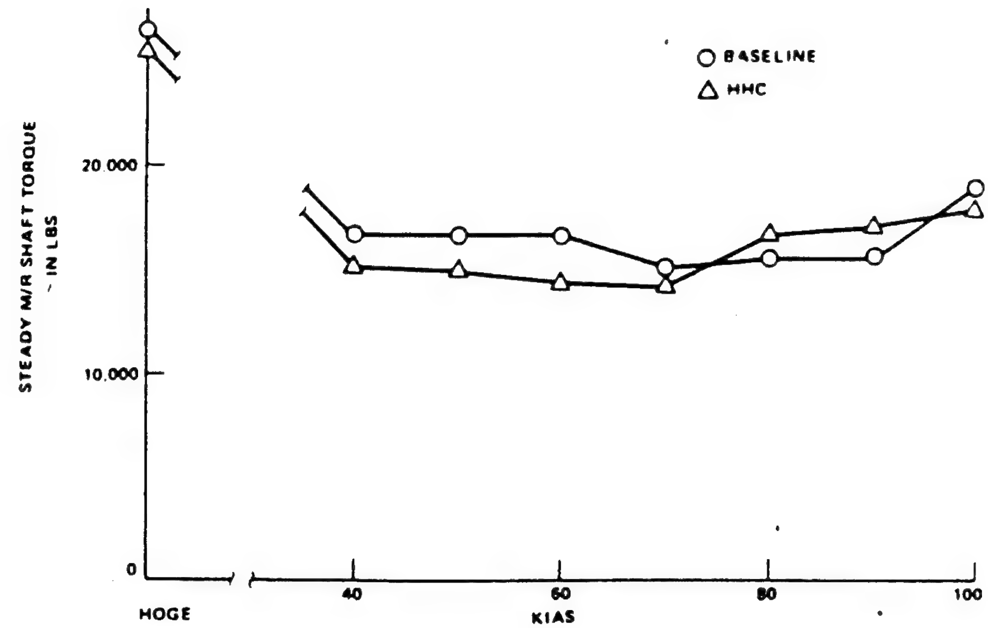

Figure 22. Main rotor torque versus airspeed with closed loop HHC (From Ref. 1)

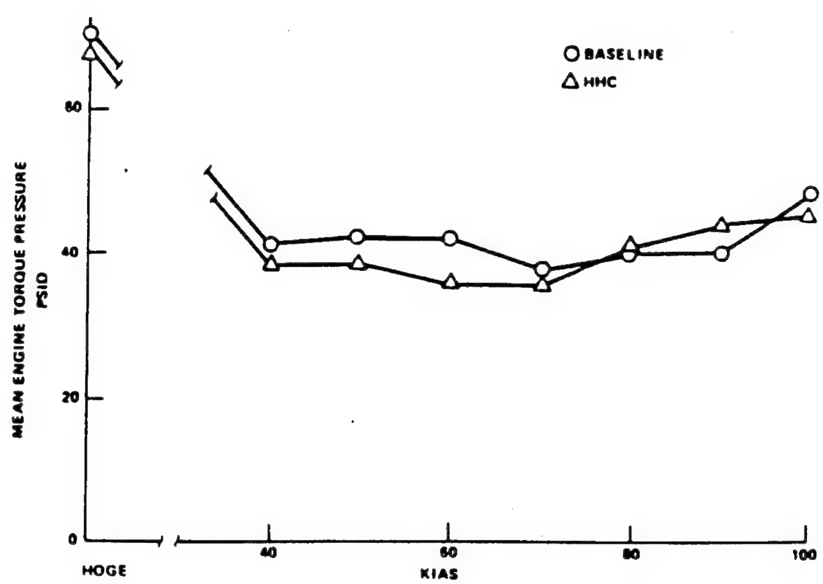

Figure 23. Engine torque pressure versus airspeed with closed loop HHC (From Ref. 1)

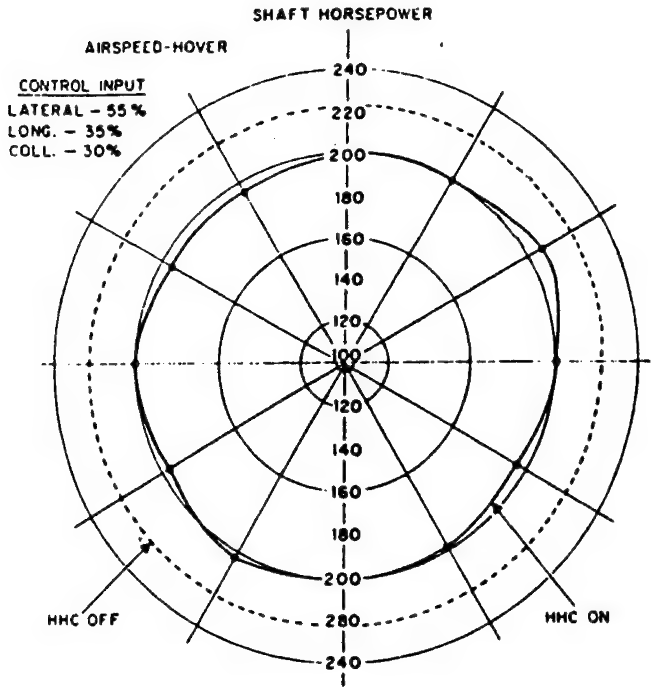

Figure 24. Effect of HHC on main rotor power at hover (From Ref. 6)

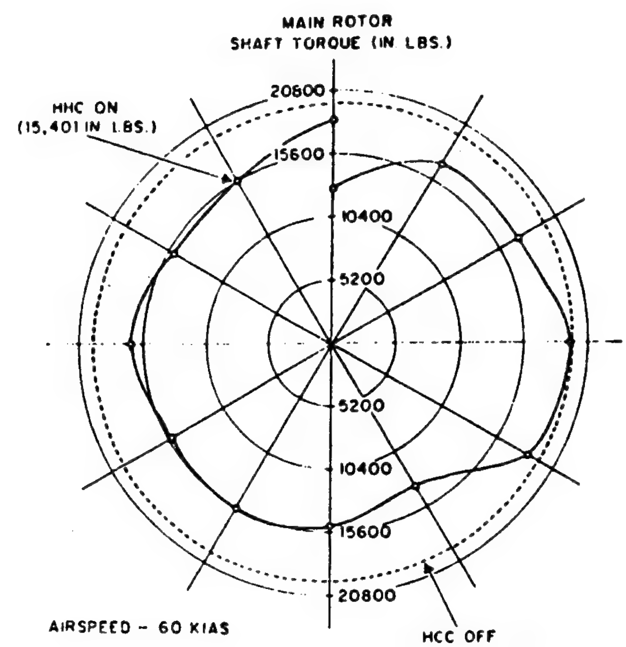

Figure 25. Effect of HHC on main rotor power at 60 KIAS with 0.33 degree lateral input. (From Ref. 6)

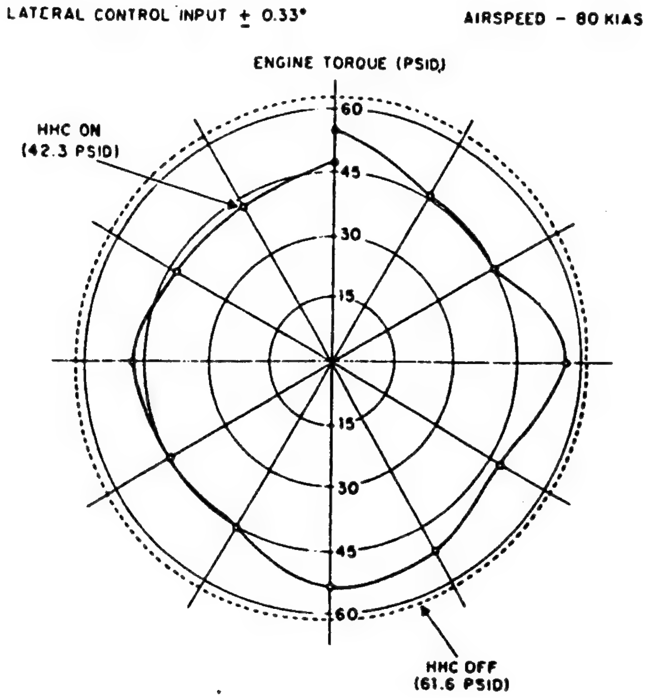

Figure 26. Effect of HHC on main rotor power at 80 KIAS with 0.33 degree lateral input. (From Ref. 6)

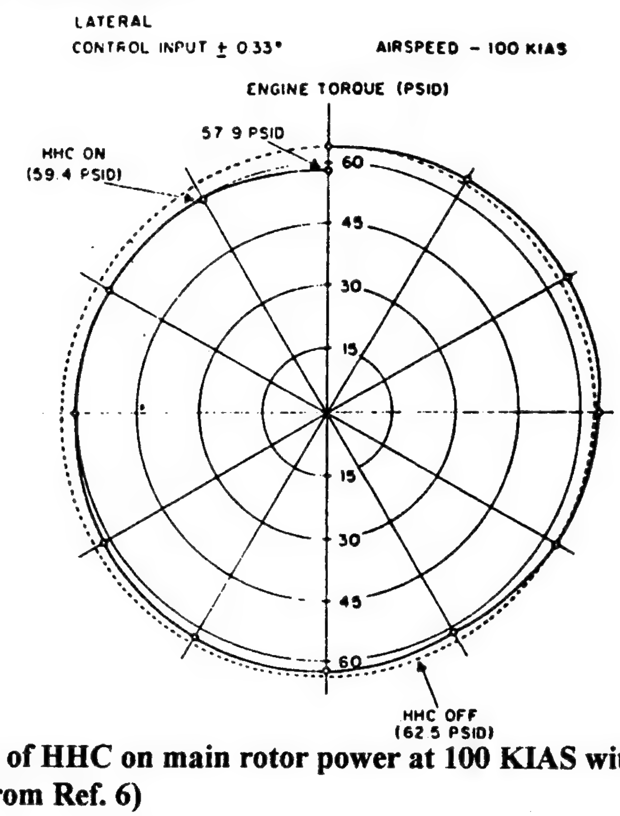

Figure 27. Effect of HHC on main rotor power at 100 KIAS with 0.33 degree lateral input. (From Ref. 6)

approximately 15% in forward flight at 60 to 80 KIAS. The power savings then begin to wash out at the higher airspeeds. Given the fact that current helicopter tactics require that helicopters to spend approximately 50% of their flight time in a hover or the low airspeed regime (10-40 KIAS), there could be sizable fuel savings associated with the implementation of HHC. The power margin also translates to increased payload and agility.

H. THE UNSTEADY AERODYNAMICS OF HHC

A recent masters degree thesis and doctoral dissertation, as well as ongoing research in the Aeronautical Engineering department of the Naval Postgraduate School, indicate that the mechanism by which HHC achieves the indicated power savings is the unsteady aerodynamics associated with HHC. This is logical considering that HHC, by its very nature, creates an unsteady flow field by inducing rotor blade pitch oscillations which in turn create plunge oscillations.

It has long been known that a purely plunging airfoil creates a propulsive force, known as the "Katzmayr effect". This is how birds and fish propel themselves. It is also known that an airfoil acting in pure pitch will typically produce drag at most values of reduced frequency. Couch and Abourahma, in References 7 and 8 respectively, showed that in the presence of layers of shed vorticity from a leading airfoil, the "Katzmayer effect" of a trailing airfoil can be greatly enhanced. In addition, they showed that with the proper phasing of the shed wakes, wake spacing and reduced frequencies, the layers of shed vorticity help create a propulsive force from the pitching motion as well, similar to the "Katzmayr effect" of a plunging airfoil.

In Reference 7 Couch modified the classic wake induced flutter theory with infinite wakes of Loewy so that it could be applied to a finite number of wakes. Figures 28 through 30, from Reference 7 show the propulsive force coefficients obtained for various wake spacings plotted against frequency ratio for pure plunge, pure pitch and coupled

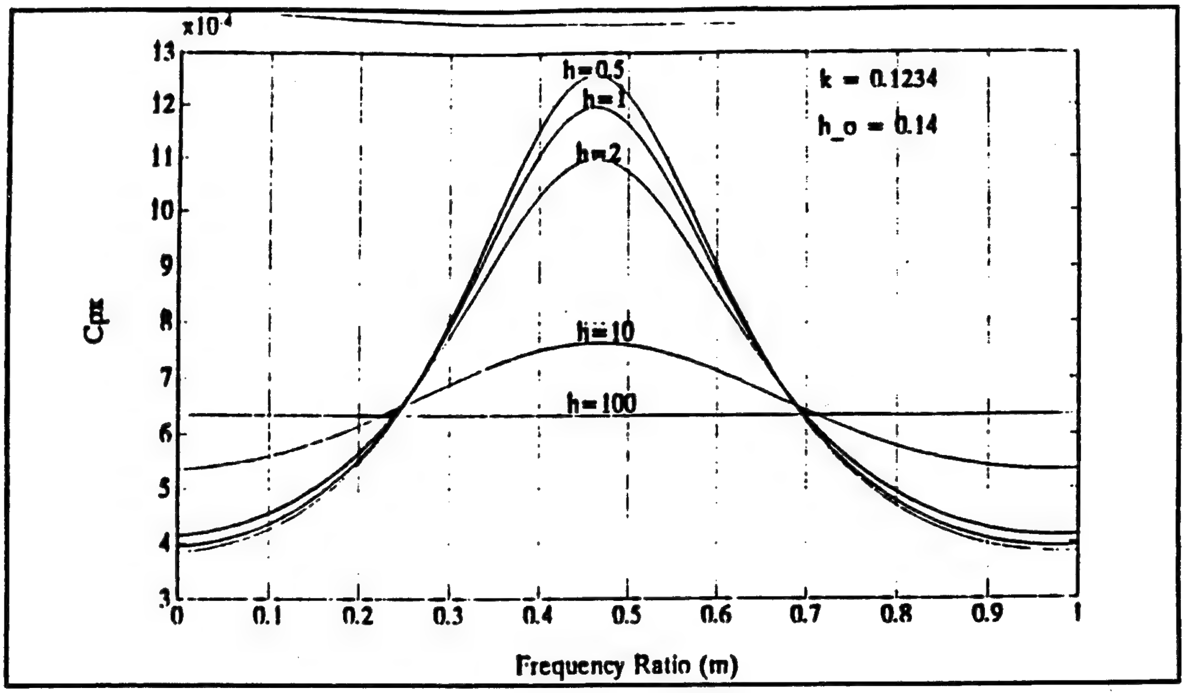

Figure 28. Propulsive force coefficient in pure plunge as a function of wake spacing with a single wake. (From Ref. 7)

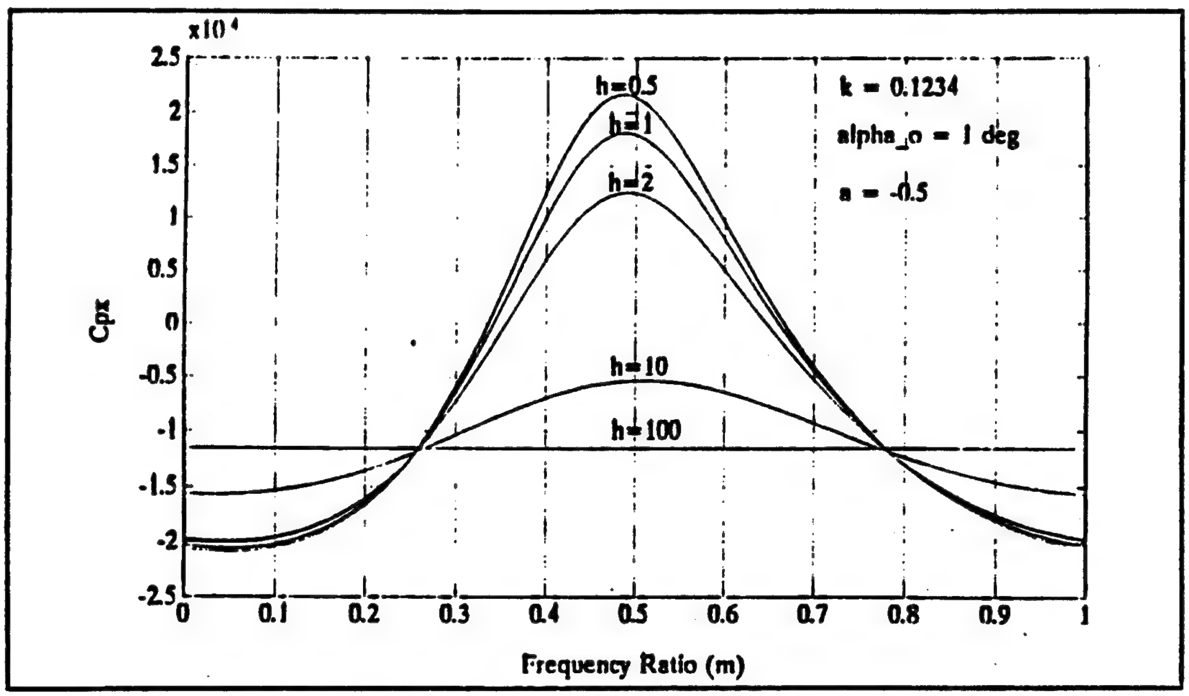

Figure 29. Propulsive force coefficient in pure pitch as a function of wake spacing with a single wake. (From Ref. 7)

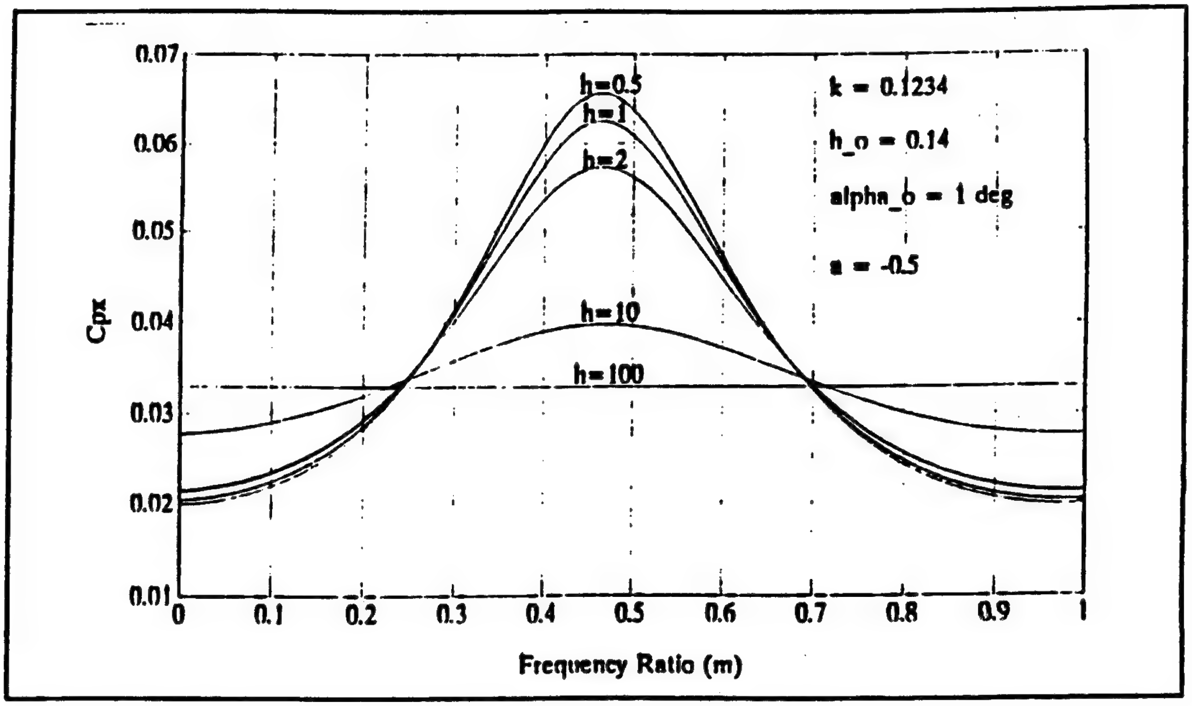

Figure 30. Propulsive force coefficient in coupled pitch-plunge as a function of wake spacing with a single wake. (From Ref. 7)

pitch-plunge for the case of a single wake. In these figures the following definitions apply:

1. C_{px} is the propulsive force coefficient
2. "m" is the ratio of the oscillation frequency to the rotational frequency.

$$m = \frac{\omega}{\Omega}$$

3. "h" is the non-dimensional distance between layers of shed vorticity.

$$h = \frac{2\pi v}{bQ\Omega}$$

where: v = freestream velocity

b = semi chord

Q = number of blades

Ω = rotational frequency of the rotor

4. "k" is the reduced frequency.

$$k = \frac{\omega b}{v}$$

5. "h_o" is the oscillating plunge amplitude.

6. "$alpha_o$" is the oscillating pitch amplitude.

7. "a" is the nondimensional elastic axis location measured from the midchord.

These plots show that there are indeed certain combinations of frequency ratios and wake spacings that create a propulsive force. The situation where m= 0.5 corresponds to two succeeding wake layers being 180 degrees out of phase with each other. Figure 31 graphically shows the vortex interaction which occurs when the wakes from a preceding airfoil and the current airfoil are 180 degrees out of phase. The logical question that is now raised is that of multiple wakes. Figure 32 shows the effect that multiple wakes have on the propulsive coefficient. Notice that as additional wakes below the rotor disk are included in the analysis, the frequency ratio for the maximum propulsive coefficient shifts. Also notice from Figures 28 through 30 and Figure 32 that as the number of wakes

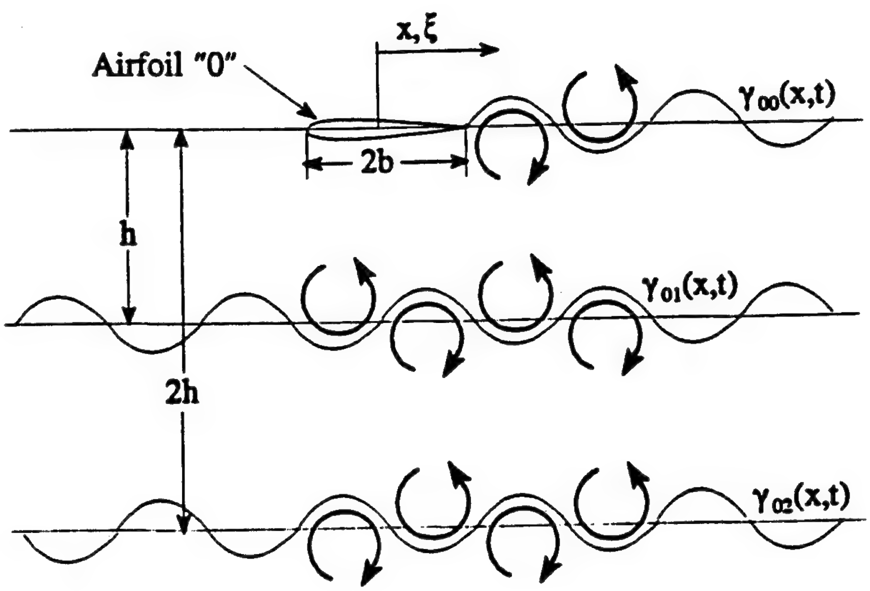

Figure 31. Vortex interaction when wakes are 180° out of phase (m = 0.5) (From Ref. 7)

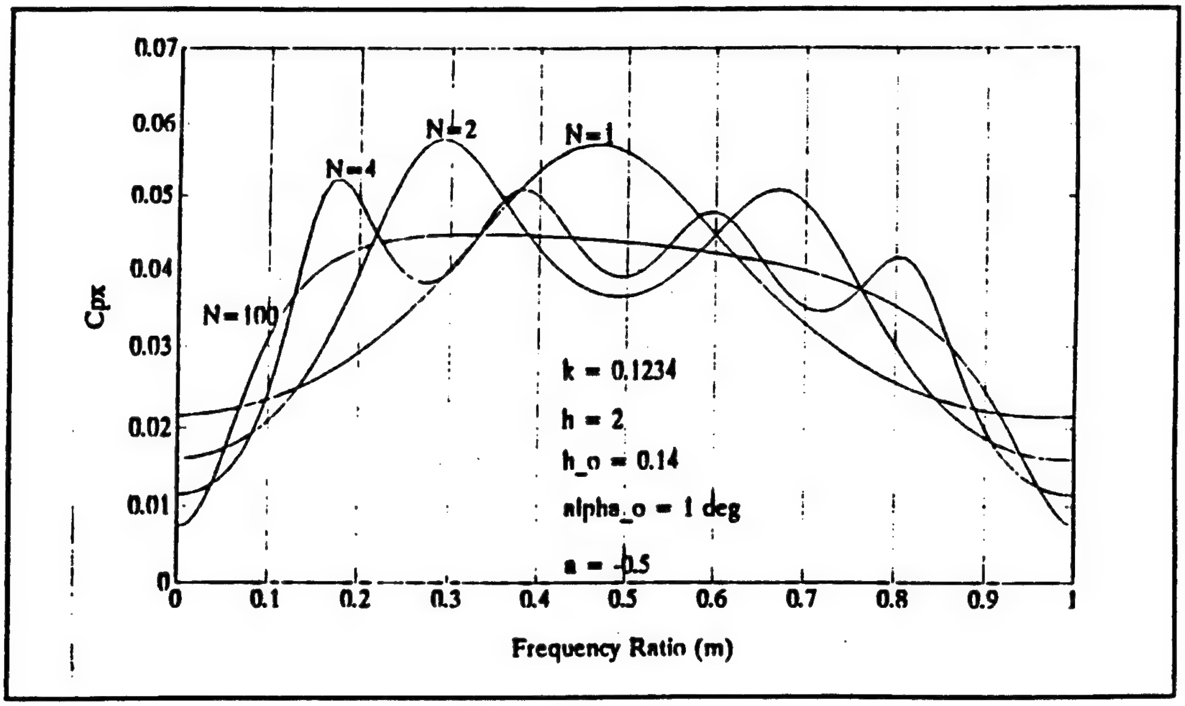

Figure 32. Propulsive force coefficient in coupled pitch-plunge as a function of the number of wakes at h = 2.0. (From Ref. 7)

considered or the wake spacing increases, the propulsive coefficient decreases. This effect can be seen in Figure 33 which is the power required curves for the OH-6A with HHC on and off. Notice that there is a power savings associated with the use of HHC throughout the speed range tested, but at either end it is considerably less than in the mid range. This is consistent with the theoretical results obtained from References 7 and 8. At the lower airspeeds and at hover the number of wakes present increases and approaches infinity. At the higher airspeeds the spacing is increased towards infinity. As explained previously, both cases result in a lower C_{px} .

In his analysis of the HHC data and in an attempt to explain the power savings associated with HHC, Couch correctly concluded that it was a result of the coupled pitch-plunge motion of the blades. However, he could not justify the amount of savings documented in the HHC data.

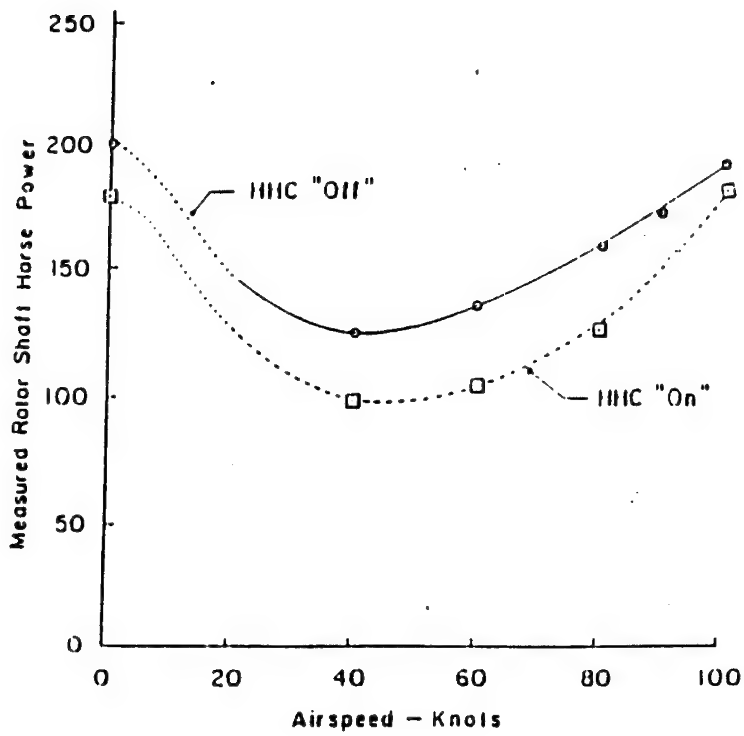

Figure 33. OH-6A power requirements with and without HHC (From Ref. 8)

In Reference 8 Abourahma goes through a lengthy and detailed analysis of the HHC data using the unsteady panel code at the Naval Postgraduate School and CFD methods. Using these tools he was able to fully justify the power savings associated with the use of HHC. He did this by continuing the analysis of Wood, et al. of Reference 9. In Reference 9 Wood, et al. divided the blade into sections and then, using Garricks analysis of a plunging airfoil, computed what the propulsive force of each section would be for a unit plunge deflection. Comparing this analysis to the actual blade deflection data from flight test and taking into account the contributions of each harmonic up through the 12th, Wood et al. were able to account for the majority of the power savings. Abourhama took into account the effects of the pitching motion as well and was able to fully justify the power savings.

I. CONCLUSIONS

In this chapter it was explained how helicopter vibrations are created and enter the airframe and how the primary objective for the NASA/Army/Hughes HHC program was to reduce these vibrations as much as possible. The theory behind HHC and its application to the OH-6A were explained in some detail. Data was shown which documents the effectiveness of HHC as a vibration control device. The theory of unsteady aerodynamics as it relates to HHC was explained and data from previous studies were presented which show that indeed a propulsive force can be generated through the coupled pitch-plunge motion of a rotor blade in the presence of layers of preceding shed wake vorticity. Finally, an explanation was offered for the power savings associated with the use of HHC during the NASA/Army/Hughes HHC flight test program of 1982. What was not presented here, but is a very important consideration, are results of acoustic testing that was conducted during the flight test program. Those data are unavailable due to their nature.

III. PROPOSED RESEARCH

A. GENERAL

Given the demonstrated benefits of HHC in the area of vibration reduction and the promising advantages in other areas as a result of its use, it is surprising that the helicopter industry has not incorporated it into existing and emerging helicopter designs. In the spring of 1994 The Honorable Mr. George Singley, Undersecretary of the Army for Research, Development and Technology, directed that a research center for helicopters be established along the same lines as the Army's National Automotive Center. The National Rotorcraft Technology Center is a cooperative effort between the four major helicopter manufacturers of this country, NASA and the FAA. Each member contributes financially to the center and votes on the most promising areas in which to conduct research in order to aid the U. S. helicopter industry. The direction from the Department of the Army is to emphasize research in the areas of vibration reduction, external noise reduction and reduction in power requirements. Since HHC has a major effect on all of these areas it is logical to once again begin intense research and experimentation in this area. However it is extremely costly to perform flight test, so a more cost efficient method of demonstrating the advantages of the use of HHC should be utilized until sufficient interest is generated to justify a flight test program. This chapter documents efforts that have been made from July, 1994 through May, 1995 to renew research efforts in this area and progress towards that research.

The intention of this research initiative is to mount a full scale, fully instrumented OH-6A main rotor system, with HHC installed, on a rotor test stand and conduct testing that will duplicate, to the extent possible, the flight testing conducted by Hughes Helicopters in the early 1980's. Since the testing will be done in an enclosed laboratory and not in a wind tunnel the testing must be restricted to hover and near hover conditions. Data will be collected in the areas of vibration reduction, rotor power requirements,

relative acoustic measurements, flow visualization of flow through a full scale rotor, and flow visualization of blade vortex interaction. This research is a cooperative effort between The Naval Postgraduate School, the United States Naval Academy and McDonnell Douglas Helicopters. Recently, SATCON Technology has expressed an interest in participating in this program to help them fulfill the requirements of two recently awarded SBIR's. Appendix A is a statement of work prepared by MDHC and delivered to SATCON delineating work items that must be accomplished enroute to completing the research program. To proceed in an orderly manner and to begin testing with a known baseline against which to compare results the HHC system to be utilized will be the same one that was used by Hughes Helicopters in 1982 -1984. If the testing proceeds as anticipated, modifications to the HHC system may be made in the future.

In order to progress with this initiative there are several objectives that must be met, the major ones are listed below. It should be remembered that as with any experimental program there are always problems that emerge, therefore this list should not be taken as comprehensive in any manner.

1. Obtain an OH-6A main rotor system
2. Instrument the rotor system
3. Locate a test facility
4. Modify and update the test facility as required
5. Design and assemble a control system for the test rotor system

What follows are the steps that have been taken towards accomplishing these objectives. Any designs for modifications included here should be considered as preliminary designs only until further progress is made to a point where the designs can be finalized.

B. ROTOR SYSTEM ACQUISITION

In this cooperative effort it is the responsibility of the Naval Postgraduate School to acquire a rotor system. Efforts to accomplish this objective have been ongoing since July, 1994. This initiative comes at opportune time in that the U. S. Army is in the process of deleting the OH-6A from their helicopter inventory. Initial contact was made with the OH-6A/ Cobra Program Office at ATCOM in St. Louis, Mo. in July of 1994. While the personnel at that facility have been pleasant to deal with and helpful to the extent possible, working through bureaucratic channels is slow and painful and often not fruitful. In March of 1995 LtGen Forester, the U.S. Army Military Deputy to the Assistant Secretary of the Army for Research, Development and Acquisition visited the NPS to receive a briefing from Dr. E. R Wood concerning the Vertical Flight Research Institute that has been established here. The General is very supportive of the efforts being made in the area of rotary wing research at the NPS. The problems encountered with the OH-6A acquisition were made known to him and he has taken steps to help ensure that NPS receives an OH-6A as the Army releases them from service. The point of contact for this acquisition is now Dr. John Johns, Chief of the Research Support Division for the U.S. Army Aeroflight Dynamics Directorate. Dr. Johns can be reached at comm. 314-263-0345. At this point it appears as though NPS will receive 2 OH-6A helicopters in the June 1995 timeframe.

The NPS could greatly benefit from having one or more helicopters in its possession. In addition to the HHC research planned for the OH-6A rotor system the remainder of the airframe will be used for dynamic modeling and analysis. They will make a superior lab tool for the purpose of designing and implementing a fly-by-wire flight control system, since there is not yet one in use, and for human factors and cockpit design studies. Having one or more helicopter fuselages will also enable the school to conduct full scale fatigue testing on tailbooms, where helicopters historically have fatigue

problems, and investigate methods of alleviating them. They will be a great asset to the Aeronautical and Mechanical Engineering Departments.

C. ROTOR SYSTEM INSTRUMENTATION

The actual testing portion of this program is being conducted with all of the care and safety considerations that would be present with a flight test program. In some ways conducting laboratory experiments of this type may be more hazardous than a flight test program. Because of this concern for safety, the rotor system and test pylon that it will be mounted on will be heavily instrumented. In addition to the safety of test parameters, there are also a considerable number of data parameters called for. While the number of parameters may seem excessive, it is easier to over instrument in the beginning and collect as much data as possible than to conduct an entire test program and then have the results disputed due to a lack of supporting data.

McDonnell Douglas Helicopter Corporation will be responsible for the instrumentation of the rotor system. An instrumentation list is included as Table 2. The list was compiled jointly by the author and Mr. Gene Munson, Director of the Controls lab at MDHC. Mr. Munson can be reached at comm. 602-891-3776. Mr. Munson was the flight test engineer for the NASA/Army/ Hughes HHC flight test program in the 1980's. The instrumentation list is based on the instrumentation utilized during the flight test program and the collective experience of Mr. Munson and the author. The data will be transmitted to the data acquisition equipment over 2 PCM streams. At this time it is undecided as to whether the data will be passed through sliprings and on to the collection equipment via wire or if it will be transmitted via a rotating mux bus, which would preclude the need for sliprings. Once the rotor system is acquired and delivered to MDHC it is estimated that approximately 3 months will be required for the instrumentation and instrumentation checkout. The system will then be shipped to the U.S. Naval Academy where the test facility is located.

Measurement #	Measurement Description	Units Measured	Bandwidth	Load Range	Comments
1000	M/R Pitch Link Load (Red) Ax F *	Lbs	0 to 250 Hz	(+/-) 1000 Lbs	
1001	M/R Pitch Link Load (White) Ax F *	Lbs	0 to 250 Hz	(+/-) 1000 Lbs	
1002	M/R Blade FWB 15% Span *	In Lbs	0 to 250 Hz	(+/-) 6000 In Lbs	
1003	M/R Blade CWB 15% Span *	In Lbs	0 to 250 Hz	(+/-) 12,000 In Lbs	
1004	M/R Blade Torsion 15% Span *	In Lbs	0 to 250 Hz	(+/-) 3,000 In Lbs	
1005	M/R Blade FWB 30% Span	In Lbs	0 to 250 Hz	(+/-) 4000 In Lbs	
1006	M/R Blade CWB 30% Span	In Lbs	0 To 250 Hz	(+/-) 10,000 In Lbs	
1007	M/R Blade FWB 50% Span *	In Lbs	0 To 250 Hz	(+/-) 4000 In Lbs	
1008	M/R Blade CWB 50% Span *	In Lbs	0 To 250 Hz	(+/-) 7500 In Lbs	
1009	M/R Blade Torsion 50% Span *	In Lbs	0 To 250 Hz	(+/-) 2000 In Lbs	
1010	M/R Blade FWB 70% Span (*)	In Lbs	0 To 250 Hz	(+/-) 3000 In Lbs	
1011	M/R Blade CWB 70% Span (*)	In Lbs	0 To 250 Hz	(+/-) 7500 In Lbs	
1012	M/R Blade Torsion 70% Span (*)	In Lbs	0 To 250 Hz	(+/-) 1000 In Lbs	
1013	M/R Pitch Housing FWB *	In Lbs	0 To 250 Hz	(+/-) 12000 In Lbs	
1014	M/R Blade FWB 90% Span *	In Lbs	0 To 250 Hz	(+/-) 2500 In Lbs	
1015	M/R Blade CWB 90% Span *	In Lbs	0 To 250 Hz	(+/-) 10000 In Lbs	
9001	M/R Blade Feathering Angle (Bl # 1) *	Degrees	0 To 100 Hz	(-10 to 25) Deg's	
9002	M/R Blade Flapping Angle (Bl #1) *	Degrees	0 To 100Hz	(-8 to 25) Deg's	
9003	M/R Blade Lead/Lag Angle (Bl #1) *	Degrees	0 To 100 Hz	(-4 to 8.5) Deg's	
1016	M/R Mast Long Mb @ W.L. 68.25 *	In Lbs	0 To 250 Hz	(+/-) 75,000 In Lbs	
1017	M/R Mast Lateral Mb @ W.L. 68.25 *	In Lbs	0 To 250 Hz	(+/-) 50,000 In Lbs	
1018	M/R Mast Long Mb @ W.L. 73.0 (*)	In Lbs	0 To 250 Hz	(+/-) 75,000 In Lbs	
1019	M/R Mast Lateral Mb @ W.L. 73.0 (*)	In Lbs	0 To 250 Hz	(+/-) 50,000 In Lbs	
1020	M/R Mast Base Long Mb	In Lbs	0 To 250 Hz	(+/-) 75,000 In Lbs	
1021	M/R Mast Base Lateral Mb	In Lbs	0 To 250 Hz	(+/-) 50,000 In Lbs	
1022	M/R Damper Load Axial Force *	Lbs	0 To 250 Hz	(-1500 to 2000) Lbs	
1023	M/R Shaft Torque #1 *	In Lbs	0 To 250 Hz	(-10,000 to 57,000) In Lbs	
1024	M/R Shaft Torque #2 (*)	In Lbs	0 To 250 Hz	(-10,000 to 57,000) In Lbs	
1101	Longitudinal Load Link Axial Force *	Lbs	0 To 250 Hz	(+/-) 2500 Lbs	
1102	Collective Control Rod Axial Force *	Lbs	0 To 250 Hz	(+/-) 150 Lbs	
1103	HHC Longitudinal Actuator Load *	Lbs	0 To 250 Hz	(+/-) 500 Lbs	
1104	HHC Left Lateral Actuator Load *	Lbs	0 To 250 Hz	(+/-) 500 Lbs	
1105	HHC Right Lateral Actuator Load *	Lbs	0 To 250 Hz	(+/-) 500 Lbs	
9004	Main Rotor Feathering Angle (Bl #3) *	Deg's	0 To 100 Hz	(+/-) 20 Deg's	
9005	Main Rotor Lead Lag Angle (Bl #3)	Deg's	0 To 100 Hz	(-4.0 to 8.5) Deg's	
9006	Main Rotor Flapping Angle (Bl #3)	Deg's	0 To 100 Hz	(-8 to 25) Deg's	
9007	Longitudinal Control Position	Percent	0 to 6 Hz	0 to 100 percent	position meters in control room also
9008	Lateral Control Position	Percent	0 to 6 Hz	0 to 100 Percent	position meters in control room also
9009	Collective Control Position	Percent	0 to 6 Hz	0 to 100 Percent	position meters in control room also
9010	HHC Left Lateral Actuator Position *	Inches	0 to 250 Hz	(-.02 to .02) Inches	
9011	HHC Right Lateral Actuator Position *	Inches	0 to 250 Hz	(-.02 to .02) Inches	
9012	HHC Longitudinal Actuator Position *	Inches	0 to 250 Hz	(-.02 to .02) Inches	
2001	Pressure Altitude	PSIA	0 to 6 Hz	(0 to 20) PSIA	

Table 2. Instrumentation List

3001	Outside Air Temperature (OAT)	Deg ~C	0 to 100 Hz	(-40 to 60) ~Deg C	
3002	Drive Motor Case Temperature	Deg ~C	0 to 6 Hz	(-20 to 100) ~Deg C	
3003	Driveshaft Ambient Temp Upper Tower	Deg ~C	0 To 6 Hz	(-20 to 100) ~Deg C	
3004	Main Rotor Mast Ambient Temp @ Base	Deg ~C	0 to 6 Hz	(-20 to 120) ~Deg C	
3005	Main Rotor Mast Ambient Temp @ W.L. 73.0	Deg ~C	0 to 6 Hz	(-20 to 120) ~Deg C	
3006	Hyd Fluid Temp Boost System In	Deg ~C	0 to 6 Hz	(-20 to 120) ~Deg C	
3007	Hyd Fluid Temp Boost System Out	Deg ~C	0 to 6 Hz	(-20 to 120) ~Deg C	
3008	Hyd Fluid Temp HHC System In	Deg ~C	0 to 6 Hz	(-20 to 120) ~Deg C	
3009	Hyd Fluid Temp HHC System Out	Deg ~C	0 to 6 Hz	(-20 to 160) ~Deg C	
5001	HHC Longitudinal Feedback Accel (*)	~G's	0 to 250 Hz	(-3.0 to 3.0) ~G's	
5002	HHC Lateral Feedback Accel (*)	~G's	0 to 250 Hz	(-3.0 to 3.0) ~G's	
5003	HHC Vertical Feedback Accel (*)	~G's	0 to 250 Hz	(-3.0 to 3.0) ~G's	
5004	Mast Mounting Plate (Whrl Twr) Vertical Accel *	~G's	0 to 250 Hz	(-3.0 to 4.0) ~G's	Bandpass for two signals Nband (0 to 50 Hz) Wband (0 to250 Hz)
5005	Mast Mounting Plate (Whrl Twr) Lateral Accel *	~G's	0 to 250 Hz	(-3.0 to 4.0) ~G's	"——————————"
5006	Mast Mounting Plate (Whrl Twr) Longitudinal Accel	~G's	0 to 250 Hz	(-3.0 to 4.0) ~G's	"——————————"
7001	Main Rotor RPM (NR) *	~%	0 to 6 Hz	(0 to 120) ~%	
7002	Main Rotor RPM (NR)	~RPM	0 to 50 Hz	(0 to 520) ~RPM	
7003	Main Rotor Azimuth Index	~Counts	0 to 50 Hz	(0 to 512) ~Counts	
7004	HHC ECU DC RPM	~Hertz	0 to 75 Hz	(0 to 50) ~Hertz	
7005	Whirl Stand Motor Torque	~In Lbs	0 to 100	(0 to 6000) ~In Lbs	
7006	Whirl Stand Motor RPM	~RPM	0 to 100 Hz	(0 to 2400) ~RPM	
7007	Whirl Stand Motor Current	~Amps	0 to 100 Hz	(0 to 400) ~Amps	Sum of the three phases of the motor
7008	Whirl Stand Motor Voltage	~Volts	0 to 100 Hz	(0 to 500) ~Volts	"——————————"
5007	M/R Mast W.L. 73.0 Vertical Acceleraton *	~G's	0 to 250 Hz	(-3.0 to 5.0) ~G's	
5008	M/R Mast W.L. 73.0 Lateral Acceleration *	~G's	0 to 250 Hz	(-3.0 to 5.0) ~G's	
5008	M/R Mast W.L.. 73.0 Longitudinal Acceleration	~G's	0 to 250 Hz	(-3.0 to 5.0) ~G's	
1106	Whirl Tower Load Cell #1 @ 45 Deg	~Lbs	0 to 50 Hz	(-1000 to 3000) ~Lbs	0 Deg Ref #1 Blade over nose
1107	Whirl Tower Load Cell #2 @ 135 Deg	~Lbs	0 to 50 Hz	(-1000 to 3000) ~Lbs	"——————————"
1108	Whirl Tower Load Cell #3 @ 225 Deg	~Lbs	0 to 50 Hz	(-1000 to 3000) ~Lbs	"——————————"
1109	Whirl Tower Load Cell #4 @ 315 Deg	~Lbs	0 to 50 Hz	(-1000 to 3000) ~Lbs	"——————————"
1110	Whirl Tower Summed Load Cells ~Total thrust	~Lbs	0 to 50 Hz	(-1000 to 5000) ~Lbs	
1111	Boost System Hydraulic Pressure In	~PSI	0 to 100 Hz	(0 to 2000) ~PSI	
1112	Boost System Hydraulic Pressure Out	~PSI	0 to 100 Hz	(0 to 750) ~PSI	
1113	HHC System Hydraulic Pressure In	~PSI	0 to 250 Hz	(0 to 4000) ~PSI	
1114	HHC System Hydraulic Pressure Out	~PSI	0 to 250 Hz	(0 to 1000) ~PSI	
8001	HHC ECU Sine Reference Output	~Sine	0 to 50 Hz	(-10 to 10) ~Sine	
8002	HHC ECU Cosine Reference Output	~Cosine	0 to 50 Hz	(-10 to 10) ~Cosine	
8003	HHC Vertical Vibration (Sine)	~Volts	0 to 50 Hz	(-2 to 2) ~Volts	
8004	HHC Lateral Vibration (Sine)	~Volts	0 to 50 Hz	(-2 to 2) ~Volts	
8005	HHC Longitudinal Vibration (Sine)	~Volts	0 to 50 Hz	(-2 to 2) ~Volts	
8006	HHC Collective Command (Sine)	~Volts	0 to 50 Hz	(-2 to 2) ~Volts	
8007	HHC Collective Command (Cos)	~Volts	0 to 50 Hz	(-2 to 2) ~Volts	
8008	HHC Lateral Command (Sin)	~Volts	0 to 50 Hz	(-3 to 3) ~Volts	

Table 2. Instrumentation List

8009	HHC Lateral Command (Cos)	~Volts	0 to 50 Hz	(-3 to 3) ~Volts	
8010	HHC Longitudinal Command (Sin)	~Volts	0 to 50 Hz	(-3 to 3) ~Volts	
8011	HHC Longitudinal Command (Cos)	~Volts	0 to 50 Hz	(-3 to 3) ~Volts	
8012	HHC Vertical Vibration (Cos)	~Volts	0 to 50 Hz	(-2 to 2) ~Volts	
8013	HHC Lateral Vibration (Cos)	~Volts	0 to 50 Hz	(-2 to 2) ~Volts	
8014	HHC Longitudinal Vibration (Cos)	~Volts	0 to 50 Hz	(-2 to 2) ~Volts	
8015	Determinant (THAT) Mantissa	~Volts	0 to 12 Hz	(-10 to 9.9951) ~Volts	
8016	Determinant (THAT) Exponent	~Volts	0 to 12 Hz	(-10 to 9.8438) ~Volts	
8017	ZO HAT (1) Exponent	~Volts	0 to 12 Hz	(-10 to 9.8438) ~Volts	
8018	Sine Reference	~Volts	0 to 75 Hz	(-10 to 10) ~Volts	
8019	Cosine Reference	~Volts	0 to 75 Hz	(-10 to 10) ~Volts	
1	Tape Recorder Sync Light (on/off)	~Counts	0 to 6 Hz	(0 or 32) ~Counts	
2	Mux Flip Buffer (Encoder) 5VDC Power	Discrete	0 to 6 Hz	(on/off) ~Discrete	
3	Mux Data Stream A Verfication *	~Counts	0 to 6Hz	(Static) ~Counts	
4	Mux Data Stream B Verification *	~Counts	0 to 6 Hz	(Static) ~Counts	
5	Mux Data Stream C Verfication *	~Counts	0 to 6 Hz	(Static) ~Counts	If Required
6	Mux Bus Data Recording Status (good/bad)	~Counts	0 to 6 Hz	(0 to 256) ~Counts	
7	Instrumentation Power Supply #1 (Excitation) *	~Volts	0 to 100 Hz	(0 to 28) ~Volts	
8	Instrumentation Power Supply #2 (Excitation)*	~Volts	0 to 100 Hz	(0 to 20) ~Volts	
9	Instrumentation Equipment Line Voltage 117VAC	~Volts	0 to 100 Hz	(0 to 150) ~Volts	

Table 2. Instrumentation List

D. ROTOR TEST FA(TY

The United States Naval Academy has a fully enclosed rotor test facility located on the ground floor of Rickover Hall. The facility is controlled by Dr. Jerry Hall of the Aeronautical Engineering Department of the USNA. Dr. Hall can be reached at DSN 281-3284 or commercial 410-293-3284. The facility, shown in Figure 34, consists of two areas, the laboratory and the control room. The laboratory contains the rotor test pylon and the adjoining control room contains the operating controls and displays as well as data acquisition equipment.

The laboratory is a chamber which is 38 feet square with 21 feet of headroom from the floor to the overhead trusses. The floor of the laboratory area is steel grating. Roll up doors on the east and south walls and a roll back roof system allow an unrestricted flow of air through the rotor system. The rotor wake passes through the grated floor and is exhausted through louvers which are located beneath the roll up doors. Bird screens are permanently installed on the roll up doors to prevent bird ingestion and a railing is built around the roll back roof to prevent person ingestion.

The test pylon, shown in Figure 35, is located in the center of the lab and is primarily a support and housing for the rotor drive mechanism. It also houses thrust pickup sensors, a photoelectric sensor which provides operating pulses to a stroboscope located on the lab wall, torque sensors, thermocouples and a slipring with terminals for up to 35 data channels. The pylon is an inverted cone of semi-monocoque construction consisting of a 0.25 in. thick aluminum skin over an aluminum frame made up of welded angles. The top of the pylon is a 1 in. thick aluminum plate. The pylon is 9.16 ft. in diameter at the base and 1.94 ft. in diameter at the top. It is 12.28 ft. tall. The pylon is mounted on a concrete base which houses the drive motor. The rotor drive system currently consists of a 50 hp DC motor with attached tachometer pickup and a tubular steel drive shaft which terminates at a mounting plate at the top of the pylon. The drive

Rotor Laboratory

A fully-instrumented rotor test facility

Operated by:
Aerospace Engineering Department
U.S. Naval Academy, Annapolis, Maryland

Figure 34. Rotor Test Facility

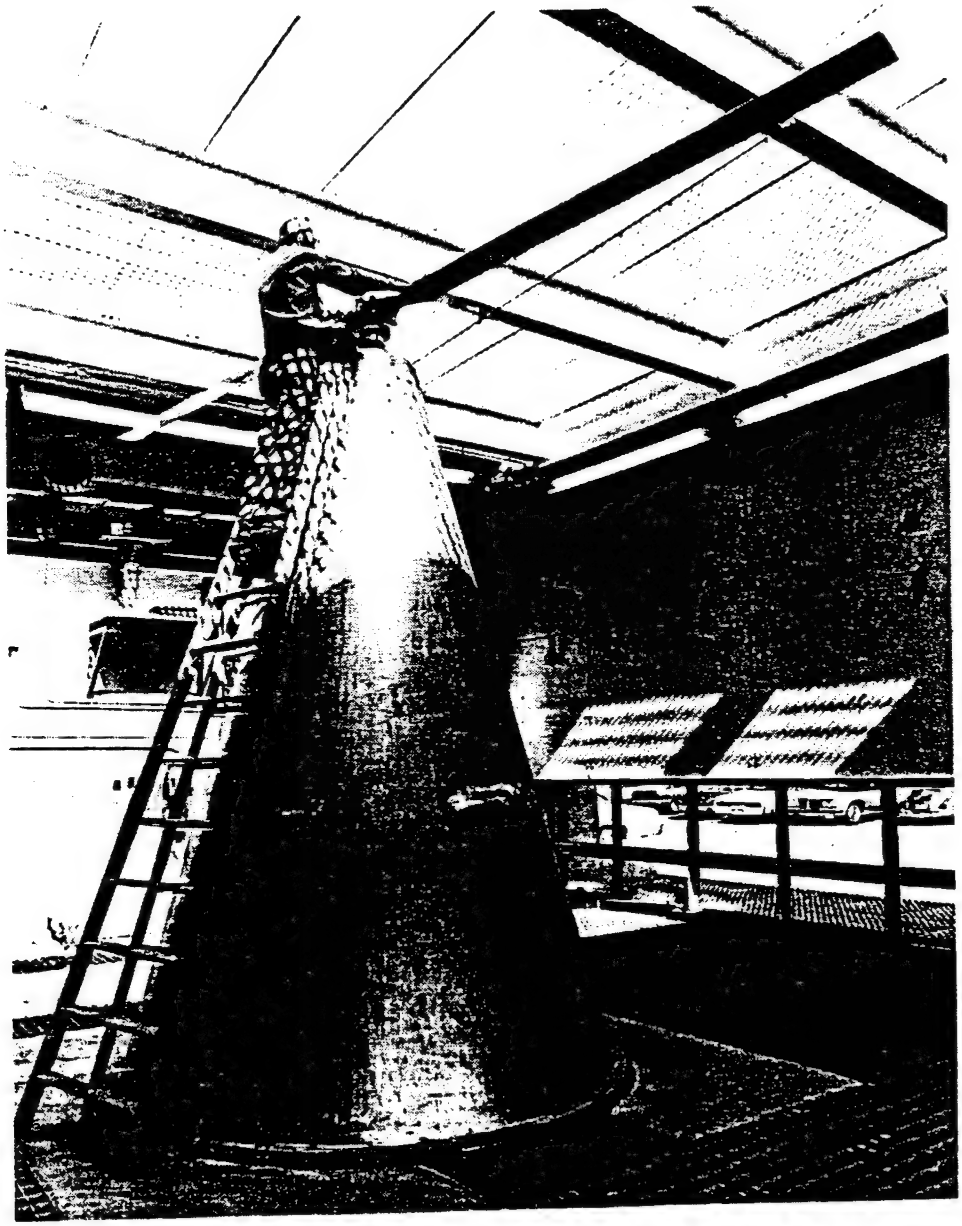

Figure 35. Rotor Test Pylon

motor is capable of infinitely variable speed from 0 to 2000 RPM with a red line limitation of 1500 RPM.

Operating controls are located in the adjacent control room. A six bay console contains the operating controls and instrumentation. The control room is located on the second floor and looks down on the lab. The rotor tip path is at approximately eye level to the observers. Bulletproof glass windows allow test personnel to observe the rotor operation. Stroboscobic lighting permits the "stopping" of blades in any position during operation. Provisions are incorporated for mounting a video camera on the rotor head to provide a view of the blades in the rotating frame of reference.

E. ROTOR FACILITY MODIFICATIONS

To accommodate the OH-6A rotor system and to gather data in the most efficient manner there are several modifications and upgrades to the existing facility that are required. Some of these modifications are specifically for the OH-6A rotor system and others are upgrades to maintain state of the art data acquisition capabilities.

In order to mount the OH-6A main rotor system on the test pylon an adapter will have to be devised that will provide secure attachment of the rotor system to the pylon, maintain the proper control rigging for collective and cyclic controls and allow sufficient accessibility to mounting bolts and the driveshaft for inspection and maintenance purposes. A coupling adapter will also be required to mate the rotating mast of the OH-6A rotor system to the driveshaft of the test pylon. The data acquisition capabilities of the facility will be upgraded with a system that will allow data acquisition and recording of two pulse code modulation (PCM) streams and real time monitoring of 32 data channels simultaneously. Finally the 50 hp motor currently installed in the pylon will be replaced by a larger one of either 100 or 250 hp.

1. Rotor to Pylon adapter

Planform views of the OH-6A main rotor system footprint and the top of the test pylon are shown in Figures 36 and 37 respectively. As can be seen from Figure 36 the attachment points of the rotor system are not symmetrical about its center in that the front feet do not project as far laterally as do the rear feet. The top of the pylon is constructed of a 1 inch aluminum ring which is welded about its entire perimeter to the sides of the pylon, which are 0.25 inch aluminum. The pylon top is further supported by 3 sets of gussets located on 120 degree centers. The gussets are welded to the underside of the pylon top as well as to the pylon sides. Concentric to the aluminum ring which forms the top of the pylon is a free floating aluminum plate which rides in a channel formed in a collar about the inner diameter of the ring. The driveshaft rides on a bearing in the center of this plate. The plate is prevented from rotating by 3 shear bolts. Connected to the underside of the plate are three load cells, which are in turn connected to the inner frame of the pylon. The OH-6A main rotor system will be mounted 6 inches above the pylon via an adapter plate that will be separated from the pylon by spacer blocks. The rotor system must be raised above the pylon for the following reasons:

1. To clear the top of the collar on the inner diameter of the pylon top.
2. To more closely align the angle the control rods make with the vertical to that of the helicopter.
3. To allow room to perform maintenance and visual inspections of the rotor system to driveshaft coupling.

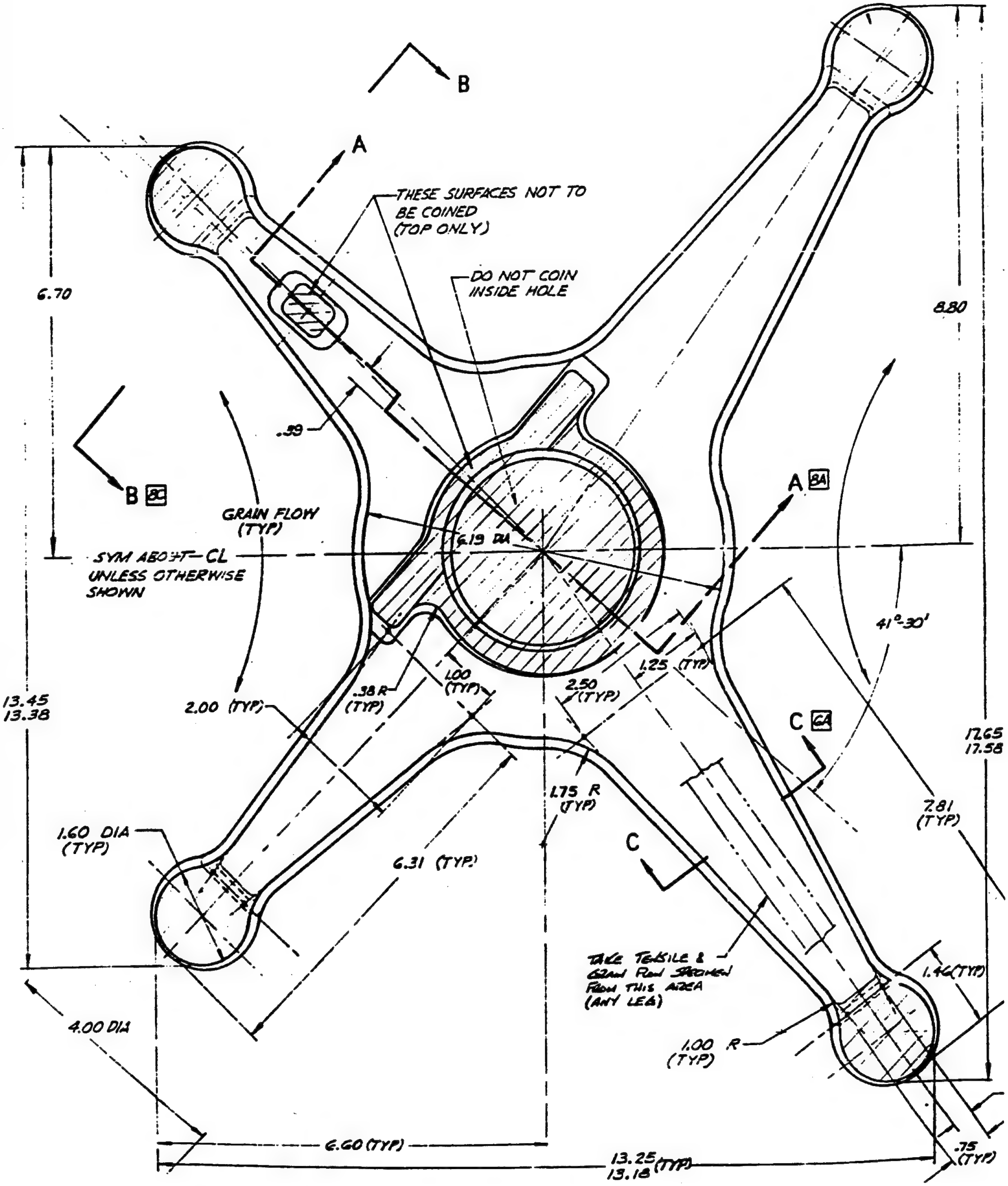

Figure 36. OH-6A Rotor Footprint

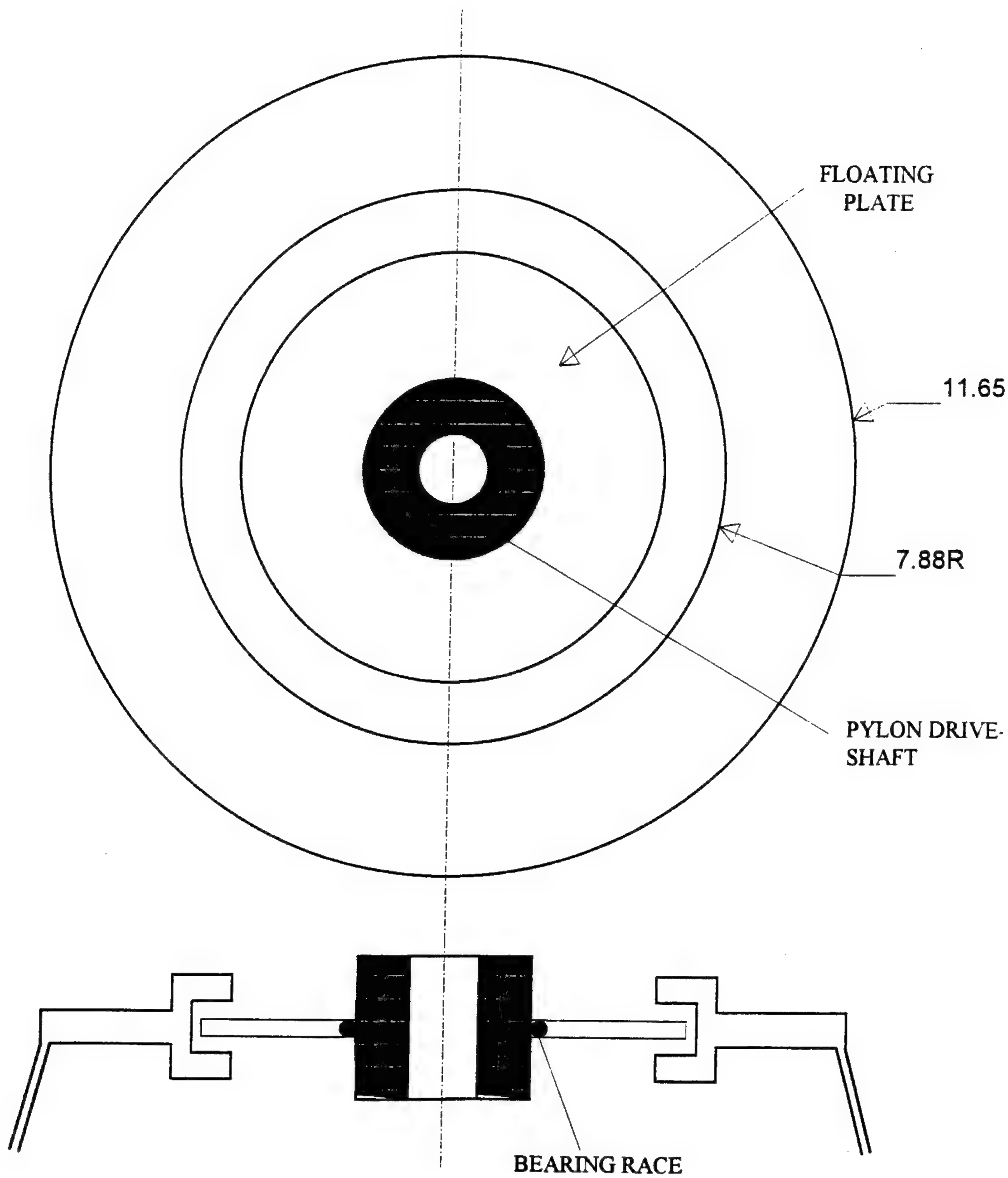

Figure 37. Rotor Test Pylon Top

The spacer blocks will be as depicted in Figure 38. They will be spaced about the top of the pylon in four locations as depicted in Figure 39. A sufficient number of blocks should be utilized so that the bottom of the adapter plate clears the pylon top by 5 inches. The actual number of spacer blocks required will depend on the material used to manufacture them. Since the blocks will be carrying very little compressive load and no tensile load it is recommended that the spacer blocks be manufactured of a high quality plywood such as Baltic or some other plywood of equal quality.

The adapter plate will be as depicted in Figure 40. It will be a ring with the same outer diameter as the top of the pylon. It is recommended that the adapter plate be made of aluminum plating. The adapter plate will be bolted to the pylon top with 0.5 inch SAE Grade 8 bolts which pass through the center of the spacer blocks. These are the same type bolts that are used to secure the rotor system to the helicopter. It is recommended that the bolts be installed so that the nuts are in the up position (on the adapter plate) and that the nuts be torque striped for easy visual inspection. Flat washers should be installed between the bolt head and the Pylon top and lock nut washers should be installed between the securing nut and the adapter plate.

The rotor system will be secured directly to the adapter plate. The mounting holes for the rotor are also depicted in Figure 40. The mounting holes for the rotor system should be oversized to allow slight position adjustments when the rotor is mounted so that the rotor driveshaft can be properly aligned with the pylon driveshaft. Oversized washers will be required to account for the oversized holes in the adapter plate. Once again, SAE Grade 8, 0.5 inch diameter bolts should be used to secure the rotor system to the adapter plate.

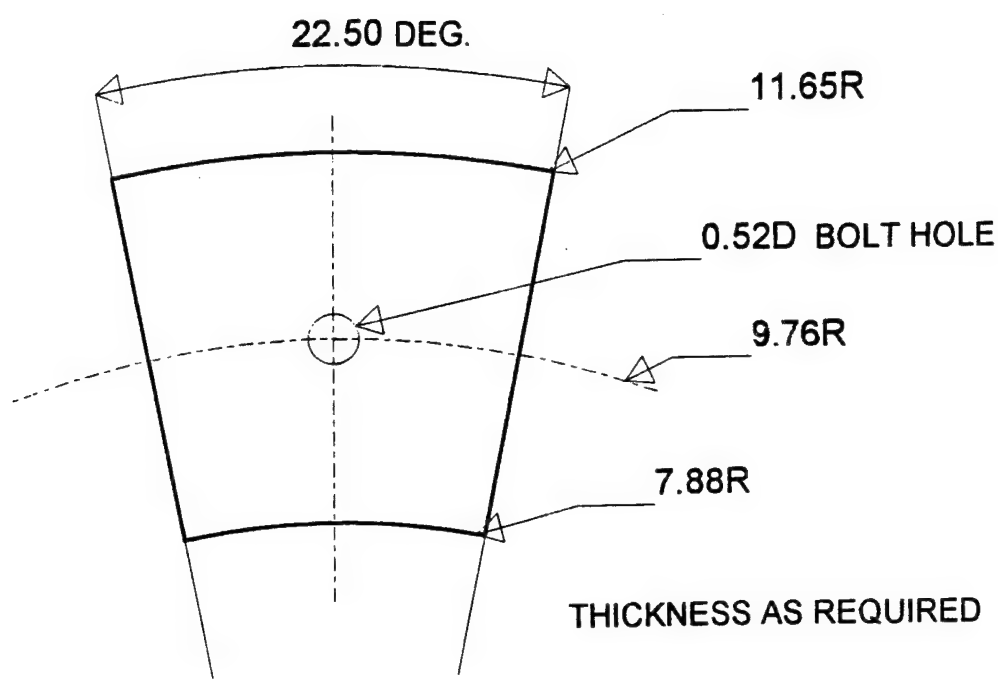

Figure 38. Spacer Blocks

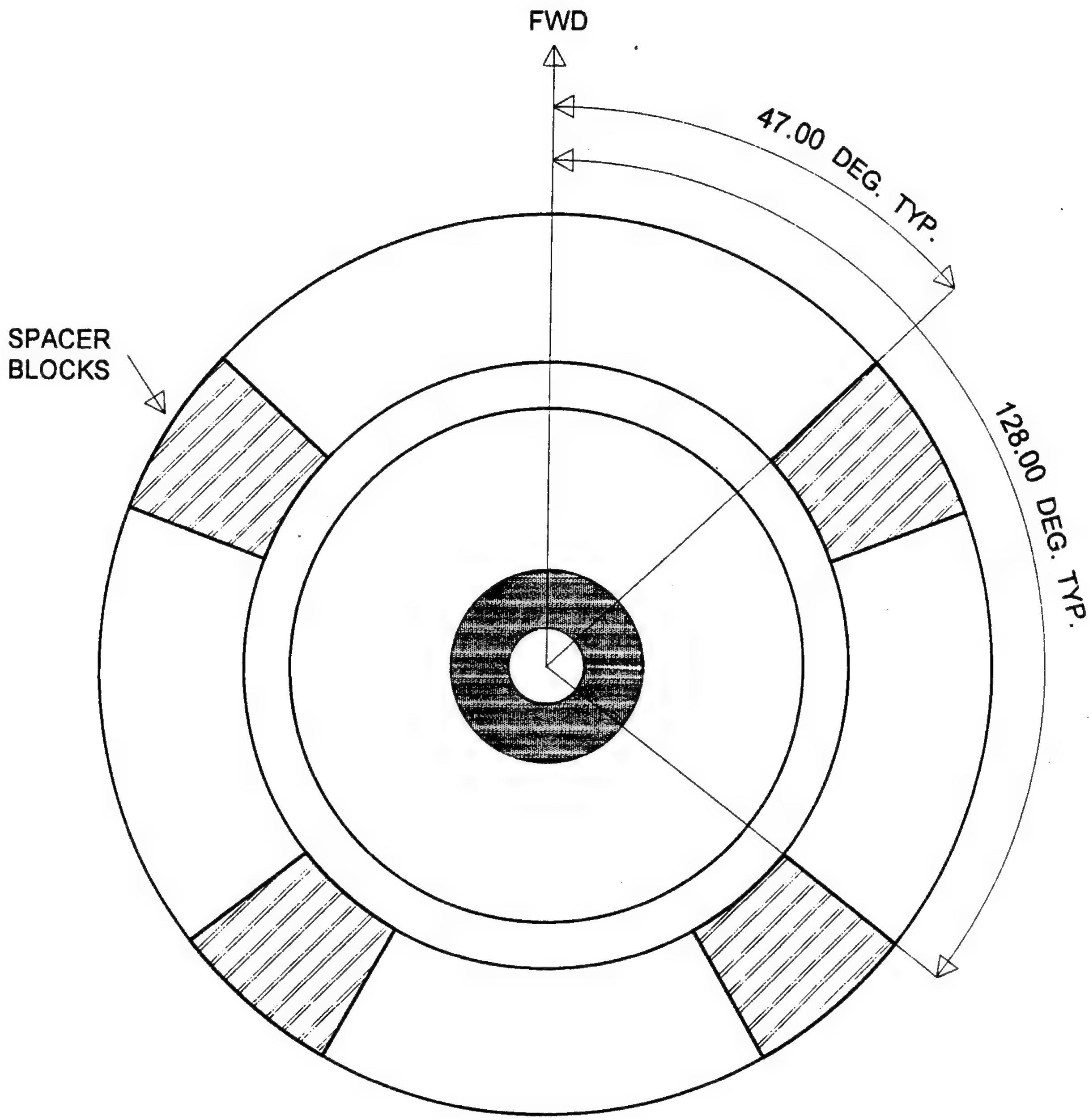

Figure 39. Spacer Block Placement

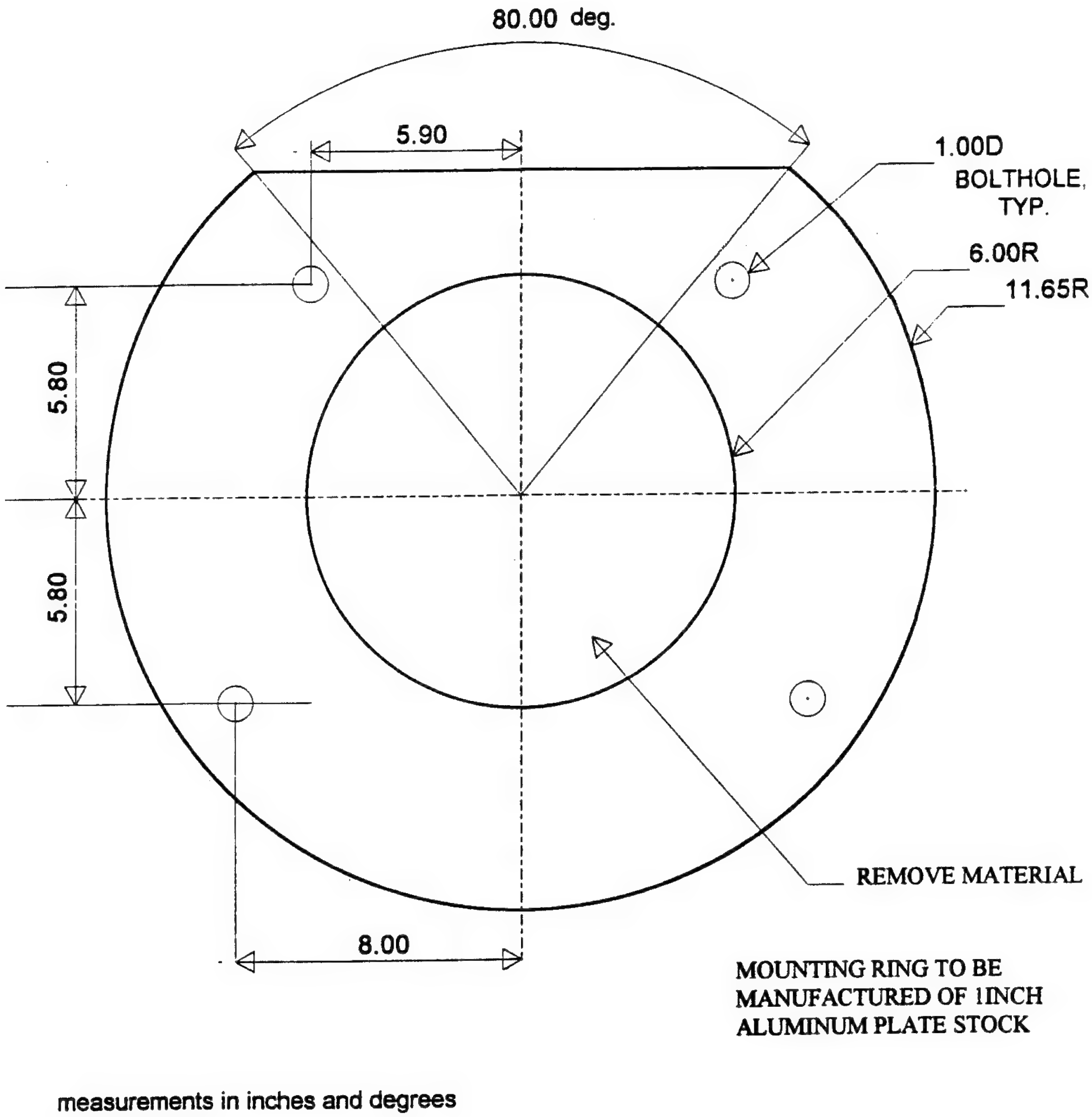

Figure 40. Adapter Plate

As previously mentioned the bolts which secure the rotor system to the aircraft are SAE Grade 8, 0.5 inch diameter bolts. To dispel any fears that these bolts may not be large enough, a brief load analysis follows:

From Mark's Standard Handbook for Mechanical Engineers the proof strength of SAE Grade 8 bolts is 120 ksi. Using a nominal cross sectional area of 0.1963 in.^2 for a 0.5 inch diameter bolt yields a capacity of 23,556 lb_f per bolt. From Reference 3 the following equation was used to estimate the amount of thrust that can be developed by an OH-6A rotor system being driven by a 250 hp motor.

$$T = \frac{38(hp)(FM)}{\sqrt{DL}} \tag{3.1}$$

T= thrust

hp= horsepower

FM= figure of merit, approximately 0.75 for the OH-6A

DL= Disk loading, approximately 4.6 psf for the OH-6A

Using the above formula, the thrust that will be generated is estimated to be approximately 3,334 lb_f. This results in a load per bolt of 833.5 lb_f. The factor of safety is then 28.26.

Since the mounting points for the rotor system are not symmetrical about its center, to ensure that the reaction per bolt would be equal, a finite element model of the rotor system was constructed using rigid bars and a 1000 pound vertical load was applied. The vertical reaction per bolt was 250 lb_f. COSMOS/M software was used for the finite element model. Even though the reactions will be slightly higher when the tip path plane is tilted, they will not be considerably higher and given the factor of safety involved, will not be a concern.

2. Rotor System Rotating Driveshaft to Pylon Driveshaft Coupling

The OH-6A helicopter utilizes a concentric stationary and rotating mast concept wherein rotational motion is transmitted to the rotor from the transmission via a rotating mast within the static mast. The static mast transmits all flight loads to the airframe and the rotating mast carries only torsional loads.

The lower end of the rotating mast protrudes below the static mast system approximately 2.25 inches and mates with the transmission. A male spline gear with a 2.22 inch major diameter is integral to the lower end of the rotating mast. The driveshaft for the test stand is a tubular steel shaft with a 5 inch outer diameter and a 2 inch inner diameter. When the rotor system is mounted atop the pylon the two shafts must mate together. It is proposed that the pylon driveshaft be modified to accept an adapter that will accommodate the rotating shaft of the rotor system. It is felt that it will be easier to manufacture an adapter and modify the pylon driveshaft to accept the adapter than to modify the rotating mast of the rotor system.

Preliminary drawings for the adapter are presented in Figure 41. It is proposed that the adapter be a cylindrical section 2.5 inches deep with a 3 inch outer diameter. The interior of the cylinder will be milled to accept the male spline fitting of the rotating shaft of the rotor system. The adapter will be secured to the pylon driveshaft by two 3/8 in. square keyways. It is recommended that the coupling and key be milled from ASTM A36 steel with a minimum yield strength of 36 ksi or from a higher grade steel.

The spline utilizes a major diameter fit. The spline data, taken from MDHC drawing number 369D25133 is given in Table 3. In order to accept this adapter, the center of the pylon driveshaft will have to routed from the current 2 inch inner diameter to 3 inches and to a depth of 2.5 inches along its longitudinal axis.

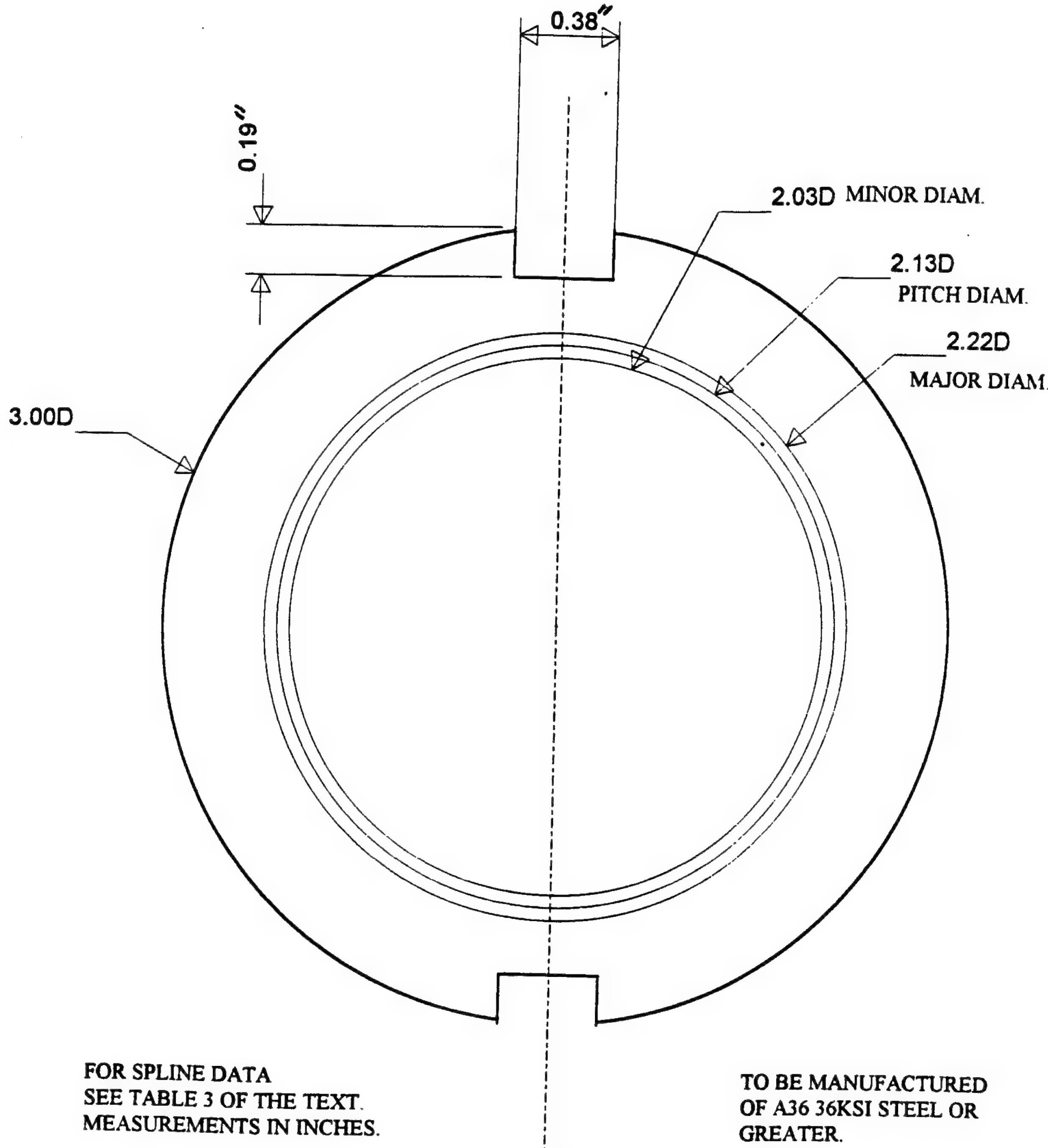

Figure 41. Shaft Coupling

Adapter Spline Data	
Number of Teeth	34
Diametral Pitch	16/21
Pressure Angle	20^0
Major Diameter	2.2200 - 2.2245 in.
Pitch Diameter Ref.	2.1250 in.
Form Diameter	2.200 in.
Minor Diameter	2.030 - 2.035 in.
Base Diameter Ref.	1.9968 in.
Root fillet radius curvature	0.005 in.
Min. effective cir. space width	0.0982 in.
Max. effective cir. space width	1.001 in. ref.
Min. actual cir. space width	0.0999 in. ref.
Max. actual cir. space width	0.1018 in.

Table 3. Adapter Spline Data

A load analysis for the key and the spline pins follows. The torque transmitted by the pylon driveshaft being driven by a 250 hp motor is computed by equation 3.2, which is found in chapter 15 of Reference 10.

$$HP = \frac{T(in-lb)xRPM}{63{,}025} \tag{3.2}$$

For the purpose of this thesis the RPM is 483 and the applied horsepower is 250. Rearranging and solving yields a torque of 32,622 in-lbs. which is carried by the pylon driveshaft. For a coupling radius of 1.5 inches the force that will be applied to the keys is: 32,622 in-lb/1.5 in. = 21,748 lbs. The required cross sectional area of the keys can be found by dividing the applied load by the allowable shear stress. From paragraph 1.5.1.2.1 of Reference 11 the allowable shear stress is 0.4 F_y, which yields 14,400 psi allowable shear stress for this application. Therefore A_{req} = 21,748 lbs/ 14,400 psi = 1.51 in.^{2}. The total cross sectional area of two 3/8 in. keys is (2)(0.373)(2.5) = 1.875 in.^{2} which is greater than 1.51 in.^{2} therefore is sufficient.

From paragraph 1.5.1.5.1 of Reference 11 the allowable bearing stress on milled surfaces is $0.9F_y$, which for this case yields an allowable bearing stress of 32,400 psi. The required bearing area of the keys is therefore (21,748 lbs./32,400 psi) = 0.67125 in.^{2} This in turn yields a required width of (2)(0.67125 in./2.5 in.^{2})= 0.52 in. The total width of two 3/8 in. keys is (2)(0.375 in.)= 0.75 in. 0.75 is greater than 0.52 therefore two 3/8 in. keys is sufficient.

Knowing the torque carried by the shaft, the total load that is carried by the coupling teeth is computed just as it was for the keys, only the radius used will be the pitch radius of the rotor system male spline, which is 2.125 in./2. The total load carried by the spline is then calculated to be 2(32,622 in.-lb)/2.125 in.= 30,703 lbs. The load carried per tooth is 30,703 lb./34 teeth= 903 lbs. per tooth.

3. Upgraded Data Acquisition System

The Data acquisition system at the Naval Academy was state of the art 20 years ago when it was installed. The current system is all analog and can accommodate only 35 data channels. LORAL Test and Information Systems of San Diego, CA has agreed to donate to the Naval Academy a LORAL Model ADS 100 data acquisition system. The ADS 100 is a state of the art, user friendly data acquisition system that can be easily tailored to meet the users needs. The system is built around a base system that is the core of the Advanced Decommutation System (ADS 100A) and the Serial Systems Analyzer (SSA 100A). The base system contains 15 card slots which hold the base system and option modules. The base system modules provide the basic platform for all system operations. User defined option modules can be added to tailor the system to the users needs for real time data acquisition, data processing, simulation, storage and distribution. A block diagram of the system is shown in Figure 42. Specific features of the system that are of particular interest to this project is that it can accommodate two

ADS100 Block Diagram for McDonnell Douglas Helicopter

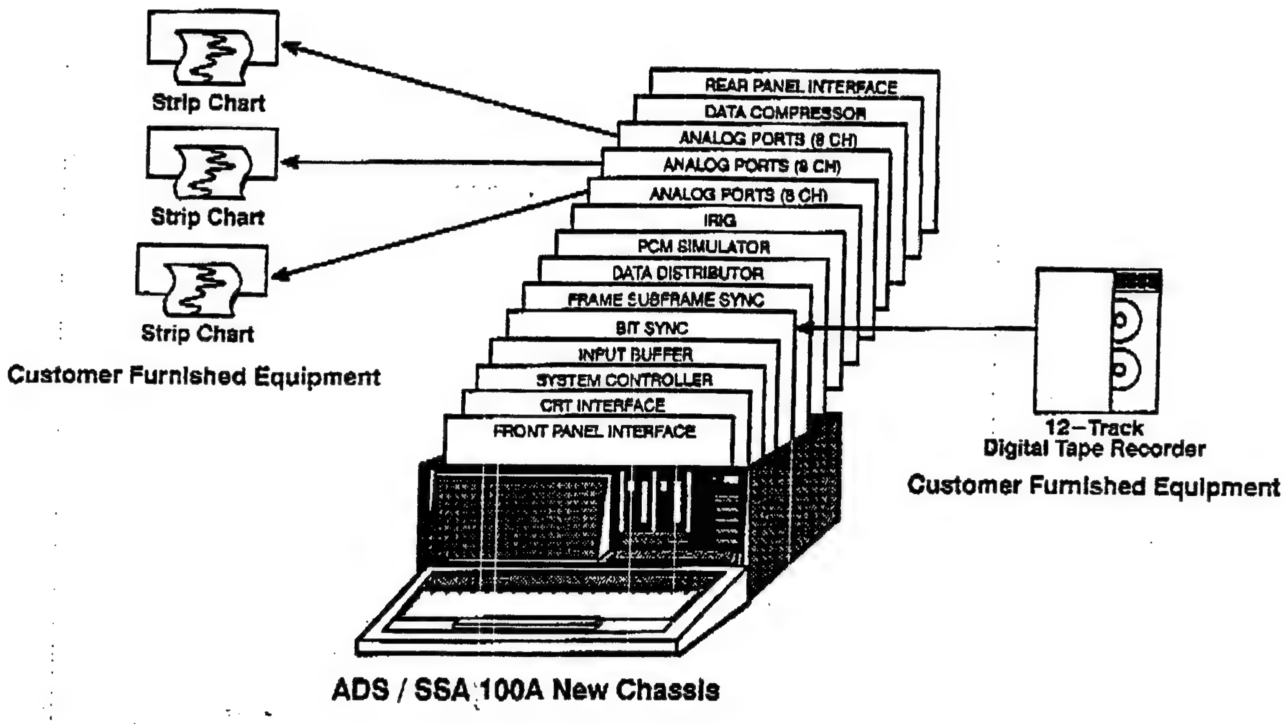

Figure 42. Data Acquisition System Block Diagram

PCM streams that can be recorded onto a 12 track digital recorder. Up to 32 of the most important channels can be decommutated and monitored real time on strip charts.

4. Upgraded Motor

The motor currently installed in the test pylon is a 50 hp dc electric motor. From Figure 33 it can be seen that 50 hp is well below the power required curve of the OH-6A. Testing a full scale rotor with a motor of this size would be fruitless. The Naval Academy has a 100 hp motor that will fit into the pylon in place of the 50 hp motor but the controller for the 50 hp motor will not handle the electrical load required by the 100 hp motor. 100 hp equates to approximately the 40 KIAS point on the OH-6A power curve with HHC on, which is the bucket of the curve. Testing with a 100 hp motor would not be ideal, but adequate data could still be obtained. It is desired to install at least a 250 hp motor and associated controller for this testing. At this time a motor has not been acquired. However, efforts are underway to acquire one.

F. HYDRAULIC AND ROTOR CONTROL SYSTEM

Hydraulic power will be required to control the basic cyclic and collective settings of the rotor as well as to operate the HHC system. The main actuators which will control the rotor system will either be the same Bertea actuators which were utilized for the hydraulic boost system of the OH-6A in the Hughes flight test or Lucas actuators of the type which are currently being used on the MD 900 Explorer. Both types of main actuator require 1500 psi for operation. The HHC actuators require 3000 psi for operation. The actuators will be connected to the rotor system by control rods with standard aircraft rod end bearings on either end.

For the purposes of this project all hydraulic power will be provided by a single, standard hydraulic maintenance cart routinely used in a fleet helicopter squadron.

Hydraulic lines will be standard aircraft braided lines. After the supply line leaves the hydraulic cart it will be split so that one pump can provide hydraulic power for both systems. Each branch will have its own accumulator and manifold for distributing hydraulic fluid to its servos. Return lines from both systems will return the fluid to a common reservoir from which the hydraulic cart will pump. A 5 micron filter will be included in the supply line to the hydraulic cart. The hydraulic fluid will leave the hydraulic cart at a pressure of 3000 psi. After the system splits into two branches, the branch that feeds the main servos will include a pressure regulator to step the pressure down to 1500 psi.

The main servos and the manifold system for the HHC system will be mounted near the top of the pylon. All other components will be securely floor mounted. McDonnell Douglas Helicopters is in possession of the required components except for the hydraulic supply cart. At this point it is too early to contact a fleet unit for use of a cart.

The rotor will be controlled from within the control room by a sidearm type controller. The controller will actually be a high grade video joy stick which will be run through a PC. Aircraft sidearm controllers are available but do not have the resolution of the video joystick. Signals from the PC will then be routed to servo amps on each main actuator. Mechanical as well as electronic stops will be utilized to limit the control travel to no more than 50% of that of the actual aircraft for cyclic control. Control position indicators will be included in line with the control system so that the operator knows the position of the tip path plane. George Lukes from the MDHC controls lab is designing this system.

The HHC system will be controlled by the hand held open loop controller that was used during the open loop flight test in 1982. MDHC, formerly Hughes Helicopters still has possession of it and it is still in working condition. Prior to its usage for this program it will be thoroughly tested and evaluated to insure its proper operation.

G. FORESEEABLE PROBLEM AREAS

There are always problems that arise during any research or experimentation program, some are foreseeable and others are not. Following are three areas associated with this program which may or may not present problems. If given prompt and correct attention these areas should not delay the conduct of the experimentation.

1. Larger Test Stand Drive Motor

The largest problem associated with a new drive motor is finding a 250 hp motor that can be mounted so that the output driveshaft is vertical and will fit within the physical constraints of the current 50 hp motor. If a motor with these characteristics cannot be found then a means of turning the output of a horizontally mounted motor 90 degrees must be devised.

2. Rotor Diameter

The diameter of the OH-6A rotor system is approximately 26.4 feet. It will fit quite easily into the rotor test lab which is 38 feet square, but there is concern about edge effects from the walls affecting the airflow through the rotor. At this time it has not been decided whether or not the rotor blades will require shortening. If they do require shortening, MDHC must be consulted on the best way to shorten the blades and whether or not blade tip weights will need to be replaced in the tips of the shortened blades.

3. HHC Feedback

The currently proposed scheme to mount the rotor system to the test pylon essentially creates a rigid connection which may not provide sufficient vibratory feedback to operate the HHC controller. If this proves to be the case, the spacers beneath the mounting ring for the rotor system will have to be replaced with a more elastic material.

H. CONCLUSIONS AND RECOMMENDATIONS

In this paper the theory of HHC has been explained and data have been presented which substantiates its merits. The results of theoretical research which help to explain the unexpected power savings associated with its operation during flight test have been presented and explained. Finally plans for future research were presented. Further research is justified in this area and it is recommended that another Thesis student follow up on this work. What follows is a brief summary of what has been accomplished and the major points of what needs to be accomplished. The Appendix is a statement of work prepared by MDHC and presented to SATCON Technology as to specific work items to be completed. It is an in-depth summary of work to be completed prior to testing. Some of the items in the list have already been completed or are in work at this time.

1. Work Accomplished Thusfar

Work accomplished thusfar towards mounting an OH-6A rotor system on the test stand at the Naval Academy is as follows:

- **a.** A rotor system has been requested from the Dept. of the Army.
- **b.** Preliminary plans for an adapter and coupling to mate the rotor system to the test stand have been made.
- **c.** An instrumentation list has been made.

d. Preliminary plans for a hydraulic system have been made.

e. Preliminary plans for a rotor control system have been made.

f. An updated data acquisition system for the Naval Academy Rotor Test Facility has been obtained.

2. Work to be Accomplished

Following is a list of the major tasks to be accomplished in preparation for the testing. A specific listing of the sub-tasks is contained in the Appendix.

a. Inspect and rework the HHC system as required.

b. Determine whether or not the rotor blades will require modification.

c. Once an OH-6A is delivered, remove the main rotor system and ship it to MDHC for instrumentation.

d. Prepare a specific test plan and matrix.

e. Assemble the rotor control system at the Naval Academy.

f. Mount the rotor system.

g. Conduct a safety review board prior to testing.

h. Conduct final inspections prior to system runup.

i. In addition to the specific items contained in Appendix A, it is imperative that a vibration analysis be conducted of the test pylon to insure that no structural modes of the pylon are excited during testing.

3. Safety Considerations

Prior to system runup a specific test plan should be prepared that includes a matrix of the testing to be performed during each test period. Prior to first system runup and subsequent to system assembly at the Naval Academy, a safety review should be conducted to review the test plan, procedures, and test matrix to insure that everyone

who participates in the testing will have a thorough understanding of the program and to insure that no safety considerations have been overlooked. A thorough system inspection should also be conducted at this time. The safety review should include personnel who are knowledgeable in the subject area but have not been involved in the planning and assemblage of the system. Testing should begin with a thorough system operational checkout and should then proceed from known conditions to unknown conditions in small increments. Prior to each test period a briefing should be held which specifically covers the data points to be conducted during that session as well as any possible malfunctions and emergency procedures. Subsequent to each test session a thorough debriefing should be conducted to document the testing that was actually conducted and to solicit feedback and input for the next test session.

APPENDIX. MDHC STATEMENT OF WORK TO SATCON TECHNOLOGY

PRELIMINARY DRAFT OF STATEMENT OF WORK FOR OH-6A ROTOR SYSTEM WITH HHC FOR TEST AT THE USNA IN SUPPORT OF SATCON TECHNOLOGY CORPORATION

Introduction:

This draft Statement Of Work covers general issues that McDonnell Douglas Helicopter Systems can support for the HHC testing at the USNA whirl tower. These are basic line items required for the cohesive formulation of the HHC test program and its resource requirements.

1.0 OH-6A Rotor System, Mast Assembly and Control System

1.1 Obtain, inspect, provide status and prepare for any required maintenance plus instrumentation for the OH-6A Main Rotor System, Static Mast/Driveshaft System and Control System.

1.2 Modify the rotor system diameter to meet USNA Whirl Tower size and wall recirculation constraints, perform structural and dynamic analysis of the rotor system to validate rotor system modifications.

1.3 Instrument the OH-6A Main Rotor System, Static Mast/Driveshaft System and Control System (Maximum and Minimum levels of instrumentation).

1.3.1 This includes strain gages, position indicators, discretes, acceleration, proximity/azimuth measurement devices and thermocouples for temperature measurement.

1.4 Design, perform stress and dynamic analysis, fabricate and perform integration of Static Mast/ Driveshaft System Adapter plate with USNA Whirl Tower Pylon.

2.0 HHC and Control System Assembly

2.1 Perform inspection, necessary rework and compliance to test performance specifications and safety criteria for primary boost/isolation and HHC actuators.

2.2 Perform inspection, operational suitability, repair/rework of the Electronic Control Unit (ECU), Acceleration/4P/16P Sensor units, HHC Control Panel, Manual (Open Loop) Controller and digital controller (MRTU Type III).

2.3 Perform HHC Electronic/Electrical System autonomous performance/validity test.

2.4 Design, fabricate and integrate cables, buffers and any required line amps to operate HHC from the whirl tower control room..

2.5 Fabricate cables from ECU to HHC actuators.

2.6 Design, fabricate, integrate and install electrical/hydraulic support systems for HHC test operations.

2.7 Design, fabricate, integrate and install cyclic and collective actuators plus the remote control system required for the control room.

3.0 System Integration

3.1 Prepare data acquisition and signal conditioning system for measurements that will be utilized for main rotor/controls, static mast assembly and the whirl tower.

3.2 Assemble Rotor, Mast assembly, mast/tower adapter plate, control system (both) and hydraulics for mechanical integration and fit checks.

3.3 Integrate Data Acquisition/Instrumentation system with rotor system hardware as well as the HHC system for system setup, measurement gain and sign convention validation.

3.4 Perform total system validation and prepare for shipment to USNA.

3.5 Deliver, install, integrate and perform system validation.

4.0 System, Safety, Test and Configuration Management Support

4.1 Provide support for preparation of:
-Test Plan/ Condition Matrix
-Operating and emergency procedures
-Safety of test review and operational safety procedures

4.2 Provide instrumentation list for total system:
-Transducers (all) specifications and calibrations
-Instrumentation signal conditioning gains, filtering characteristics and bandwidth
-Calibration procedures, techniques, validation and sign convention validation requirements.

4.2 Provide operating limits and procedures for the Rotor and HHC system as required for limits and safety.

4.3 Provide and acquire spare parts as needed for on hand items for test integrity as well as items that may be required during the normal conduct of testing

4.4 Provide data acquisition/instrumentation, data reduction/analysis, interim and final report preparation operations and support.

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